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NAVAL TEST PILOT SCHOOL FLIGHT TEST MANUAL

FIXED WING PERFORMANCE

THEORY AND FLIGHT TEST TECHNIQUES

NAVAL AIR TEST CENTER
PATUXENT RIVER, MARYLAND

Mrs. Crumbocker:

Attached is the Manual, "Fixed Wing Performance Theory and Flight Test Techniques" which we discussed on the telephone. The distribution sheet is inserted as first page. Please send us AD Number on this revised copy dated July 1977.

Thank you for all your help.

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This manual is a guide for pilots, naval flight officers, and engineers attending the U. S. Naval Test Pilot School. The text presents basic theory, test techniques, and data reduction methods for fixed-wing aircraft performance testing. The scope of each flight investigation presented is limited to that necessary to teach the principles of the test and its associated data reduction methods. Students are encouraged to be alert, original, and where possible, make appropriate modifications to general

test techniques to obtain a specific result more efficiently. All flight testing described in this manual is conducted to define engine and airframe performance characteristics for use in determining mission suitability and specification compliance where applicable.

FIXED WING PERFORMANCE

THEORY

AND

FLIGHT TEST TECHNIQUES

A

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PERFORMANCE FLIGHT TESTS AND DATA REDUCTION

I. INTRODUCTION

II. PERFORMANCE FLIGHT TEST CONDITIONS AND PILOT TECHNIQUES

III. PERFORMANCE DEMONSTRATION FLIGHT

IV. PITOT STATIC SYSTEM TESTING

V. STALL AIRSPEED DETERMINATION

VI. DETERMINATION OF EXCESS POWER CHARACTERISTICS

VII. LEVEL FLIGHT PERFORMANCE

A. JET AIRPLANE LEVEL FLIGHT PERFORMANCE

B. DETERMINATION OF STANDARD DAY MAXIMUM LEVEL FLIGHT AIRSPEED

VIII. MANEUVERING PERFORMANCE

IX. CLIMB PERFORMANCE

X. DESCENT PERFORMANCE

XI. STOL PERFORMANCE

XII. DETERMINATION OF CLIMB SCHEDULES FROM ACCELERATION RUN DATA

XIII. COMPUTER DATA REDUCTION

XIV. TPS AIRPLANE DETAILS

XV. COCKPIT EVALUATION

APPENDIXES

REVISIONS

Insert latest changes, destroying superseded pages.

SECTION	PAGES AFFECTED	DATE OF CHANGE

1. __________
2. __________
3. __________
4. __________
5. __________
6. __________
7. __________
8. __________
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PREFACE

"Consideration of the human variables involved in flight leads me to the conclusion that the pilot must be as wise in decision as a good chess player and as correct in his tempo of transaction as a bullfighter."

L. I. KELLY
Aeronautics

The U.S. Naval Test Pilot School Performance Testing Manual has been prepared as a guide for the student who is learning the principles of performance flight testing. The flight techniques discussed herein are many of those which are used in performance flight test projects. The principles illustrated provide a basis for modifying these techniques to fit a specific performance test problem.

The acquisition of data and data analysis are fundamental to performance flight testing. Frequently it is a very laborious process. At the U.S. Naval Test Pilot School, the quantity of data obtained and the data analysis methods used are limited to those necessary to teach the principles. Though the data analysis methods presented in this manual are valid and are perfectly suitable for some purposes, more refined and costly methods exist.

SECTION I

INTRODUCTION

PURPOSE OF THE PERFORMANCE TESTING MANUAL

This manual is a guide for pilots and engineers attending the U.S. Naval Test Pilot School (USNTPS). The text presents basic theory, test techniques, and data reduction methods for fixed-wing performance testing. The scope of each flight investigation presented is limited to that necessary to teach the principles of the test and its associated data reduction methods. Students are encouraged to be alert, original, and where possible, make appropriate modifications to general test techniques to obtain a specific result more efficiently. All flight testing described in this manual is conducted to define engine and airframe performance characteristics for use in determining mission suitability and specification compliance where applicable.

FLIGHT TEST SYLLABUS

The performance flight test syllabus is conducted concurrently with the academic course on airplane performance. A theoretical analysis of fixed-wing engine and airframe performance is presented in the academic syllabus. In addition, flight briefings on test techniques and "Scope of Test" are conducted prior to each flight. Equipped with a knowledge of theory and test techniques, the student plans, conducts, and reports on the assigned test flight.

Each fixed-wing study group will be assigned an airplane for the specific purpose of determining its engine and airframe performance characteristics. The syllabus flights flown by the group to accomplish this evaluation are typical of those conducted at the Naval Air Test Center (NATC) during Navy Preliminary Evaluations (NPE), Board of Inspection and Survey (BIS) Trials, and Performance Technical Evaluations (PTE). Many of the tests require several similar flights to

satisfactorily complete the scope of testing desired. For these tests, the various flights will be divided among the group pilots.

Each student should prepare an informal written report for all performance exercises as soon as possible after each flight in order to record the pilot's freshest impressions and interpretations of the test results. The report must be brief when discussing satisfactory characteristics and as detailed as necessary to describe particularly good and bad characteristics. As a minimum, it should include quantitative data on properly annotated rough plots, qualitative comments, and any airplane peculiarities. Upon completion, the report will be presented either as an informal debrief or as a formal NATC written report. In any case, each individual report will be incorporated into a project notebook which will provide a running record of flight results for the entire group.

When all performance flights have been completed, the data from each individual report will be used to construct composite graphs of the various performance parameters. These graphs enable the group to evaluate the mission suitability of the airplane and to determine specification compliance. At the end of the performance syllabus, the group compiles data and information gathered during the course into a formal oral report similar to a PTE. Early in the performance syllabus, the various tests must be put in perspective with a view toward their inclusion in this final report.

SCHOOL APPROACH TO PERFORMANCE TESTING

The Test Pilot School (TPS) provides a modern airplane for performance testing; and although the airplane is not a new one, the school approach is to assume that it has never been evaluated by the Navy. The test project assumes no

NPE was conducted and that BIS had designated TPS to conduct the secondary phase Aircraft and Engine Performance Trials and Technical Evaluation. The airplane is assumed to be designated for present day use. Stability and control, weapons delivery, and other testing is assumed to be assigned to other divisions of NATC. The TPS student is charged with the responsibility of determining and reporting on the engine and airframe characteristics of the project airplane.

"Mission suitability" is an important phase at NATC, and its importance is reflected in the whole theme of flight testing at TPS. The fact that an airplane meets the requirements of pertinent Military Specifications may be of secondary importance if any performance characteristic degrades the airplane's normal operation in its planned environment. The mission of each airplane will be discussed, then students must conclude whether or not the performance characteristics they evaluate are suitable for the intended mission. This conclusion must be supported by a logical discussion and analysis of quantitative and qualitative observations. Since students possess and can communicate recent fleet experience, their conclusions should be based on a "users" concept.

The performance of an airplane cannot be evaluted for comparison with contract guarantees or other airplanes without accurate quantitative data. Consequently, accurate numbers are indispensable in providing the student with a foundation on which to build a logical and analytical analysis. At TPS every effort is made to test under ideal weather conditions and with all sensitive instrumentation operative, but problems in either of these areas may cause slight errors in the data. If the errors are not the result of improper test techniques, the test results will show the desired data trends which are essential for a satisfactory report. If bad weather, instrumentation failure, or poor pilot test technique result in large

magnitude errors or improper trends, critique on the data may be given; and if warranted, the flight will be reflown. It is important to remember that the primary purpose of the performance testing syllabus at TPS is teaching proper flight test techniques and the basic ideas supporting the methods.

The fact that TPS does not require the data to be exactly correct before it is presented in a report does not excuse the student from conducting a critical examination of the data. It is important for the student to know that errors in the final data do exist and that he should explain the effect of the errors on the results. It has been the experience of the school faculty that some students are too critical of the accuracy of their data because of the scatter they observe in their graphs. Many factors enter into the degree of scatter observed in performance data. High among these factors are inconsistent pilot technique and instrumentation errors. Often a certain amount of scatter is to be expected depending on the particular test under consideration. Again, a degradation in accuracy of results can also be attributed to poor pilot technique, which may in turn be traced to a student who did not properly prepare for a flight or did not follow instructions. Immediate flight planning following a flight brief is imperative to ensure that the proper and necessary preparation is accomplished. The flight brief will encompass the data accuracy which is expected. If these limits are not achieved, an explanation in the individual flight report is warranted.

SECTION II

PERFORMANCE FLIGHT TEST CONDITIONS AND PILOT TECHNIQUES

PERFORMANCE FLIGHT TEST CONDITIONS AND PILOT TECHNIQUES

PURPOSE

The purpose of this section is to discuss performance flight test techniques which will be used in conducting the tests outlined in this manual. A demonstration flight, discussed in Section III, will be given to each student by a TPS flight instructor.

DISCUSSION AND THEORY

General

There are three basic test conditions under which a pilot will operate an airplane while conducting performance testing. Each test condition requires special flight techniques and utilizes different primary flight instruments for pilot reference. These conditions are stable equilibrium, unstable equilibrium, and nonequilibrium. Equilibrium test conditions are present during tests in which the airplane is stabilized at a constant airspeed and altitude. A stable equilibrium condition is a condition in which the airplane, if disturbed, will return to its initial condition. An unstable equilibrium test point is a point from which the airplane, if disturbed, will continue to diverge. A nonequilibrium test is a test during which there is a change in airspeed and/or altitude.

Stable Equilibrium Conditions

A stable equilibrium test point represents a condition at which pressure altitude, thrust (power), and flight path angle are constant and airplane acceleration along each axis is zero. Stable equilibrium data points are obtained in

both level and turning flight when operating at airspeeds greater than the airspeed for minimum drag (stable portion of the thrust or power required curve). The test technique for obtaining stable equilibrium data is to adjust altitude first, power second, and then wait until the airplane stabilizes at the equilibrium flight airspeed. It is important to point out that altitude must be maintained precisely at the desired test level and that thrust/power must not be changed once set. If this technique is followed, a time history of airspeed can be used to determine when the equilibrium data point has been obtained. For most tests, when the airplane has changed less than 2 kt in the preceding 1 minute period, an equilibrium data point can be assumed to have been achieved. Stable equilibrium test conditions are obtained most rapidly by approaching them with excess airspeed. This approach ensures convergence, whereas an accelerating approach may converge only after fuel exhaustion. The flight test technique used in obtaining stable equilibrium conditions is called the constant altitude method in most flight test literature.

The primary parameters for pilot reference when obtaining data points under stable equilibrium conditions are altitude, vertical speed, heading for straight flight, and bank angle for turning flight. There is no substitute for a good visual horizon. In airplanes equipped with automatic flight control systems (AFCS) that incorporate attitude, altitude and heading hold modes, stable equilibrium data points can be obtained by using these modes provided the sensitivity of the AFCS is adequate for the test. In straight flight, stable equilibrium conditions can be achieved by using altitude and heading hold modes. In turning flight, stable equilibrium conditions can be achieved by using altitude and attitude hold modes.

Unstable Equilibrium Conditions

Unstable equilibrium data points are more difficult to obtain but can be obtained rapidly if proper technique is employed. For the unstable equilibrium data points, indicated airspeed is held constant. Altitude, engine RPM, and bank angle may be adjusted as required by the test being conducted. Since the indicated airspeed must remain constant, it becomes apparent that good test techniques will assist in obtaining good unstable equilibrium data points. Unstable equilibrium data points are associated with the unstable portion of the thrust or power required curve. To obtain data points under these conditions, the desired test airspeed is established first, then the throttle is adjusted to climb or descend to the desired test altitude. The vertical speed indicator is an important instrument in achieving equilibrium conditions. With throttle set, the vertical speed is stabilized while maintaining the desired test airspeed. A throttle correction is made and the stabilized new vertical speed is observed. The approximate RPM required for level flight can be determined by correlating the values. For example, while attempting to obtain a level flight data point at 135 KIAS, it is determined that 88% RPM produces an 800 fpm climb in the vicinity of the desired test altitude and an 80% RPM produces a 200 fpm descent. The test pilot can determine that 1% RPM change represents a 125 fpm change in vertical speed. By adjusting throttle to 81.6%, he should achieve equilibrium level flight conditions. Normal pilot technique usually puts one within 1% or 2% of the proper RPM, but this averaging technique then applies. A variation of this technique must be used in turning flight when the throttle is set at MIL and cannot be used as the adjustable variable. In this case, bank angle (or load factor) may be related to vertical speed in the same manner

that RPM was related to vertical speed in the straight flight conditions. The flight test technique used in obtaining unstable equilibrium conditions is called the constant airspeed method in most flight test literature.

The primary parameters for pilot reference when obtaining data points under unstable equilibrium conditions are airspeed, vertical speed, heading for straight flight, and bank angle for turning flight. In tests in which rate of climb can be corrected to thrust or power required, it will not be necessary to achieve equilibrium at zero vertical speed. A small altitude change over a small time period (usually 5 min) can be used to correct the test results to level conditions. In other tests, it will be necessary to achieve zero vertical speed. A small amount of practice usually results in satisfactory ability to obtain zero vertical speed at a desired test altitude in less time than it takes to determine an average rate-of-climb correction. It should also be recognized that the constant airspeed techniques can be used to good advantage by a proficient test pilot when obtaining test data under stable equilibrium conditions. Normally, automatic flight control systems offer little advantage over manual control in obtaining unstable equilibrium data points. If control stick steering is available, this mode can be useful if the unstable region of the thrust required curve is also an unstable angle of attack region of flight. An airplane exhibits positive angle of attack stability if it initially tends to return to the same angle of attack after a sudden disturbance. If the airplane appears to exhibit positive or neutral angle of attack stability, manual control will probably be preferable. Altitude hold mode used in conjunction with throttle adjustment will seldom give satisfactory results in unstable thrust required flight conditions.

Nonequilibrium Test Points

Nonequilibrium test points are usually the most difficult to obtain. They preclude the pilot's having stable conditions or being able to trim to maintain constant conditions. The pilot does, however, have some schedule which he can follow and which he can use to assist him in correcting to achieve a satisfactory flight path or flight test condition. Some nonequilibrium tests such as acceleration runs are performed at a desired constant altitude. Others such as climbs and descents are performed according to a desired airspeed schedule.

The primary reference parameters for nonequilibrium tests will be dictated by the specific test being performed. Automatic flight control system equipment can be a great aid in obtaining nonequilibrium condition data. The degree to which it can be employed will depend upon the specific test and the ability of the modes to perform their design functions. Good heading hold and altitude hold modes can be extremely valuable in obtaining level acceleration test data. If the altitude mode is unable to maintain a relatively constant altitude ($\pm$100 ft), use of control stick steering may be superior to manual control since this mode will provide automatic longitudinal retrim. Climb and descent tests can be performed using Mach or Indicated Airspeed hold modes if they are available and if they have sufficiently high gain to maintain the desired schedule accuracy ($\pm$5 kt or $\pm$0.01M). If these modes are not available, use of control stick steering will at least provide smooth automatic retrim and relieve the pilot of that task.

SECTION III

PERFORMANCE DEMONSTRATION FLIGHT

PERFORMANCE DEMONSTRATION FLIGHT

INTRODUCTION

The performance demonstration flight conducted at TPS was established to introduce the student to the fixed-wing performance testing syllabus. The flight will usually be conducted in a different airplane than the student will utilize during the performance phase of the syllabus. Prior to the demonstration flight, the student must have completed the handbook exam, cockpit checkout, and at least one familiarization flight. In addition to test techniques, the student will be instructed in the proper use of instrumentation and the importance of knowing the correct preflight procedures and preparation required prior to each test flight.

PURPOSE OF DEMONSTRATION FLIGHT

The primary purpose of the fixed-wing performance demonstration flight is to acquaint the student with the test techniques and method of tests utilized at TPS. The knowledge gained during this flight will enable the student to practice the correct techniques during succeeding familiarization and practice flights. The purpose of the demonstration flight is to teach test techniques and not fixed-wing performance theory. The importance of the various tests and the significance of the test results relating to mission suitability will be emphasized. Methods of test and pilot techniques will be demonstrated for the following tests:

a. Takeoff and climb performance.

b. Level flight performance (thrust required).

c. Constant altitude acceleration runs (specific excess power).

d. Level turning performance and buffet boundary characteristics (thrust and lift boundaries).

e. Stall airspeed determination.

f. Pitot - static system calibration (tower fly-by).

FLIGHT FORMAT

a. The student will be briefed on the proper operation and utilization of photopanel and phenomenon lights and the proper flight test techniques.

b. The importance of altimeter settings (29.92), sufficient leader on photopanel film and frame counter correlation will be discussed.

c. The flight will be conducted simulating a fully-instrumented airplane (student in front seat).

Takeoff Techniques (Nonequilibrium)

The demonstration pilot will emphasize the importance of initial line-up and trim settings in takeoff performance testing. The use of brakes can seriously affect the takeoff distance and acceleration performance of the airplane. The technique to obtain good nose wheel line-up is uniform, simultaneous braking while lining up prior to takeoff. This will ensure that the nose wheel is not cocked at brake release. If nose wheel steering is available it may be used. Photopanel data are normally taken during the takeoff run. The photopanel is started prior to brake release, usually after takeoff thrust has been obtained. Brakes are released at a specified photopanel counter number and the phenomenon light is used to mark the film at that time. Takeoff rotation will be at a specified airspeed, gear and flaps will be retracted at a specified airspeed, and the climb schedule will be intercepted at as low an altitude as is feasible for normal operating conditions. During acceleration to initial climb airspeed, rate of climb should not exceed 500 fpm.

Climb Techniques (Nonequilibrium)

The instructor pilot will demonstrate the transition to the climb schedule, pilot techniques for maintaining schedule, and the proper use of test instrumentation. The climb schedule represents a nonequilibrium flight condition in which the pilot is following a predetermined airspeed vs. altitude schedule. The schedule for the particular airplane will be described during the flight briefing. The demonstration should cover techniques for transitioning from a constant altitude condition to a climb schedule and should include demonstration of an interrupted climb. The instructor pilot should emphasize the requirement for making smooth, early corrections during the climb phase. In demonstrating an interrupted climb, it is important to emphasize that the conditions at the time the climb is interrupted must be duplicated exactly when re-establishing the climb. At least 1,000 ft of the climb schedule should overlap.

Level Flight Performance

Stable equilibrium data points (points at airspeeds greater than the airspeed for minimum drag) will be demonstrated in level flight at the altitude designated for the particular demonstration airplane. If the test airplane is equipped with a pointer counter altimeter, it is essential that even 1,000 ft altitudes not be used in the demonstration because of the inherent altimeter instrument friction characteristics about the 12 o'clock position. The proper technique is to set the appropriate power or thrust setting first and allow the airspeed to decelerate and stabilize. After setting the thrust and stabilizing the altitude, the pilot's stopwatch should be started to determine airspeed acceleration or deceleration over a 1 min interval. When accelerations or decelerations are less then 2 kt/min, equilibrium

conditions can be assumed to have been achieved and the photopanel and/or cockpit data can be recorded. It is essential that the student understand this time history concept in order to minimize test time requirements. This is called the constant altitude method. The primary reference instruments are the altimeter, vertical speed indicator, and RMI for heading.

Unstable equilibrium data points (points at airspeeds less than the airspeed for minimum drag) will be demonstrated in level flight at the same altitude as the stable equilibrium points. This technique is called the constant airspeed method. The instructor pilot will emphasize that the airspeed must be maintained constant for unstable equilibrium data points. Pilot techniques for determining thrust or power required for level flight under these conditions will be demonstrated.

Trim the airplane at the desired airspeed and set a thrust (RPM) which results in a stabilized climb of less than 1,000 fpm at the trim airspeed. Reduce the thrust (RPM) to obtain a stabilized descent at the same trim airspeed. After determining the rate of change of vertical speed with RPM, an estimate of RPM required for level flight can be made. A throttle correction to position the airplane at the desired test altitude will then be made as the airplane approaches within $\pm$100 ft of the specified test altitude. After the airplane has maintained level, stabilized flight for approximately 60 sec, data can be recorded. An altitude gain or loss of 25 ft in 60 sec is considered acceptable. This method is applicable in airplanes which have no significant change in gross weight while obtaining unstable equilibrium data test points. The primary instruments for this technique are the airspeed indicator, the vertical speed indicator, the altimeter, and the RMI for heading.

Acceleration Run (Nonequilibrium)

The constant altitude acceleration run is another example of a nonequilibrium flight test condition. In this case, a constant altitude will be maintained, but the airplane will not be in equilibrium condition (except at V_{mrt}) because it is accelerating along the longitudinal axis. Level flight acceleration runs may be performed by constantly trimming or by attempting to hold all forces with the control stick. The trimming method is recommended; however, exact trim conditions should not be attempted. A small amount of pull or push force should be maintained throughout the run. The reason for this is to avoid the control system break-out forces about the force trim points. This condition may result in a force reversal from push to pull during the run which generally produces an oscillatory motion during the test. If the airplane has large excess thrust capabilities or slow engine spool-up time, it will be necessary to commence the acceleration run from climbing flight conditions to conduct the entire acceleration run at stable engine operating conditions. The instructor pilot should emphasize the importance of making smooth corrections and discuss the use of speed brakes to facilitate the start.

Level Turning Performance

The techniques for determining level turning performance (both stable and unstable equilibrium data points) will be demonstrated at the designated test altitude. The instructor pilot should emphasize that V_{mrt} is a point in turning performance (1.0g point). For the stable equilibrium points, set the thrust/power at MIL, establish the airplane in a constant bank, stabilize the altitude with normal acceleration and allow the airspeed to stabilize. Stabilization can be assumed when

the airspeed, normal acceleration, and altitude are stabilized for 5 sec within 1,000 ft of test altitude. The instructor should emphasize the importance of trimming the airplane longitudinally to maintain the desired bank angle and normal acceleration, and of using the visual horizon. For unstable equilibrium points, advance the thrust to MIL and adjust normal acceleration in a turn to maintain desired airspeed. Once the airspeed is stabilized by applying the appropriate normal acceleration, the bank angle required for level flight can be established. The instructor should demonstrate the relationship of bank angle to vertical speed, and of normal acceleration to airspeed. The stabilization criteria is the same as for stable equilibrium conditions. This technique is difficult to master and will require considerable practice for proficiency.

Buffet Boundary (Nonequilibrium)

The technique for determining the buffet boundary (lift limit) characteristics of an airplane will be demonstrated at the same test altitude as level turning performance. The technique is called the constant Mach number/airspeed wind-up turn and is another example of a nonequilibrium test condition. Establish the airplane at a specified Mach number/airspeed and smoothly increase normal acceleration. In order to minimize scatter in the data, a constant thrust setting is preferred. For load factors above the maximum level flight sustained g-available, use bank angle (airplane weight vector) to supplement thrust in order to maintain a constant Mach number/airspeed during the maneuver. Satisfactory data can be obtained within 1,000 ft of test altitude. The instructor should emphasize the importance of increasing the normal acceleration at 0.5g per sec or less to avoid dynamic errors in the data. The instructor should point out and discuss the applicable buffet levels.

Stall Speed Determination (Nonequilibrium)

In this test, the pilot is following a deceleration schedule of 1/2 kt/sec or less. The configuration of the airplane or engine limitations will determine the throttle setting. The primary instruments in this particular maneuver will be the airspeed indicator and the clock. Stalls at higher rates of deceleration (5 to 10 kt/sec) will also be demonstrated to illustrate the effects of nonsteady flow conditions. The instructor will also discuss or demonstrate the effects of trim setting on stall speed data.

Phototheodolite Tower Fly-by

The instructor will familiarize the student with the local course (currently in use), and the student may perform practice pitot static source calibration runs if time and fuel permit.

Student Participation

The student should be permitted to try each basic technique, but the instructor pilot must maintain the pace of the flight in order to complete all demonstration requirements. It is important to emphasize that the purpose of this flight is to introduce the three basic flight test conditions and the techniques used by test pilots to collect performance data. Instructors will answer questions, but avoid specific details of theory which may not yet have been introduced in the academic syllabus. Further detailed briefs and practice flights are included in the syllabus.

SECTION IV

PITOT STATIC SYSTEM TESTING

SECTION IV

NOTATIONS INTRODUCED IN THIS SECTION

λ	-	lag error constant
P_s	-	static pressure
P_a	-	ambiemt pressure
P_T	-	free stream total pressure
ρ_a	-	ambient density
ρ_{SL}	-	sea level density
γ	-	ratio of specific heats
q_c	-	impact pressure
q_{c_i}	-	indicated impact pressure
V_i	-	indicated airspeed
V_c	-	calibrated airspeed
ΔP	-	difference in static source pressure and ambient pressure, $P_s - P_a$
$\frac{\Delta P}{q_{c_i}}$	-	static source error coefficient
Subscript w	-	signifies data corrected to standard airplane weight
H_p	-	actual pressure altitude of the test airplane
H_{p_o}	-	observed pressure altitude of the test airplane (cockpit or photopanel recording)
$H_{p\ theodolite}$	-	presure altitude of theodolite
h_{tape}	-	tapeline height increment of test airplane above MSL
h_p	-	pressure height of the test airplane above MSL

W	-	airplane gross weight
W_s	-	standard airplane gross weight
$\Delta H_{p_{ic}}$	-	altimeter instrument error correction
ΔV_{ic}	-	airspeed indicator instrument error correction
ΔV_{pos}	-	airspeed position error correction
$\Delta H_{p_{pos}}$	-	altimeter position error correction

REFERENCES

Section IV

1. AGARD, Chapters 1 and 10.

2. NACA, Report 919.

3. Military Specification, MIL-I-6115A.

4. USNTPS - FTM - No. 102, Section III.

5. Petersen, Aircraft and Engine Performance.

6. NASA, Report TN-D-822.

7. NATC Technical Memo, TM75-2SA, 19 Nov 1975.

PITOT STATIC SYSTEM TESTING

INTRODUCTION

The altimeter, airspeed, and machmeter indicators are three universal flight instruments which require total and/or static pressure inputs to function. It is generally assumed that a properly designed pitot tube can provide error free total pressure measurements. However, it is difficult to accurately measure ambient static pressure. The error which results from the difference between the actual ambient pressure and the static pressure measured at the aircraft static pressure source is called position error. In addition, any pitot/static pressure system is subject to errors associated with both the mechanical instrument and the tubing which connects the instruments to the static pressure source. This section will briefly discuss these errors and the various test techniques used for their determination. The pertinent requirements of military specification MIL-F-6115A will be presented as well as a brief discussion of the recovery factor for an outside air temperature (OAT) probe.

PURPOSE

The purpose of this test is to evaluate the altimeter and airspeed position errors of an airplane using the altimeter depression method. In addition, the OAT probe recovery factor will be determined.

DISCUSSION AND THEORY

General

The errors associated with any pitot/static pressure system may be generally classified as mechanical, operational, and installation errors. Mechanical errors primarily include instrument errors. Operational errors include readability and incorrect barometric settings. Installation errors primarily consist of lag errors and position errors.

Instrument Errors

Instrument errors are the result of manufacturing discrepancies, hysteresis, temperature changes, friction, and inertia of moving parts. A laboratory calibration of all flight instruments must be accomplished to determine instrument errors prior to an inflight determination of position errors. Sensitive instruments may require daily calibration. When the readings of two pressure altimeters are used to determine the error in a pressure sensing system, a precautionary check of calibration correlations is often advisable. The problem arises from the fact that two calibrated instruments, placed side by side with their reading corrected by use of calibration charts, do not always provide the same resultant calibrated altitude. Tests such as the "Tower Fly-By" or the "Trailing Bomb" for altimeter calibration require an altimeter to provide a base line (reference) pressure altitude. These tests normally require that the reference altimeter be placed next to the service system altimeter prior to and after each flight. Each altimeter reading should be recorded, and if after calibration corrections have been applied there still exists a discrepancy between the two readings, this discrepancy should be incorporated in the data reduction. This correction is not made at TPS.

Lag Error

The presence of lag error in pressure measurements is generally associated with climbing/descending or accelerating/decelerating flight and is most evident in static systems. When changing ambient pressures are involved, as in climbing and descending flight, the speed of pressure propagation and the pressure drop associated with flow through a tube introduces lag between the indicated and actual pressure being measured. The pressure lag error is basically a result of the following:

a. Pressure drop in the tubing caused by viscous friction.

b. Inertia of the air mass in the tubing.

c. Instrument inertia and viscous and kinetic friction.

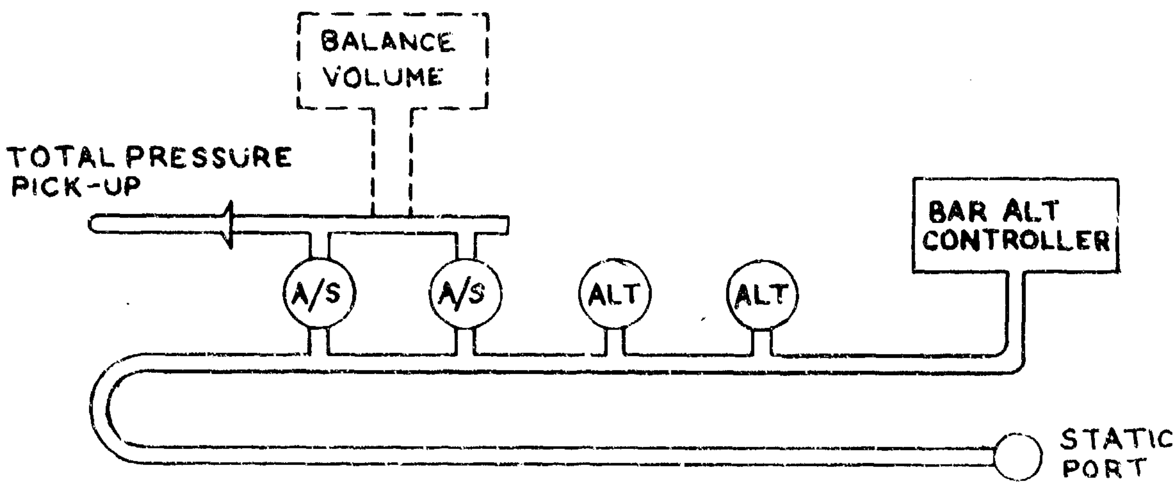
d. The finite speed of pressure propagation; i.e., acoustic lag.

Over a small pressure range, the pressure lag is small and can be determined as a constant (λ). Once a lag error constant is determined, a correction can be applied through mathematical manipulations. Another approach, which is more suitable for testing at the Naval Air Test Center (NATC), is to balance the pressure systems by equalizing their volumes. Balancing minimizes or removes lag error as a factor in airspeed data reduction for flight at a constant airspeed. This is particularly important for airplanes used as calibrated pacers.

Lag Constant Test

The pitot and static pressure systems of a given airplane supply pressures to an unequal number of instruments and require different lengths of tubing for pressure transmission. The volume of the instrument cases plus the volume of the tubing,

when added together for each pressure system, produce a volume mismatch between systems. Figure 1 illustrates a configuration in which both the length of tubing and total instrument case volumes are unequal. If an increment of pressure were applied simultaneously across the total and static sources of Figure 1, the two systems would require different periods of time to stabilize at the new pressure level and a momentary error (lag error) in indicated airspeed and indicated altitude would result. The time required for each system to stabilize can be related to the system time constant.



SYSTEM	LENGTH OF 3/16" ID TUBE	TOTAL VOLUME OF INSTRUMENT CASES
Static	18 ft	370×10^{-4} cu ft
Pitot	6 ft	20×10^{-4} cu ft

Figure 1
Analysis of Pitot and Static Systems Construction

If an increment of pressure is instantaneously applied to a system, the lag constant (λ) represents the period of time required for the pressure of the system to increase by an amount equal to 63.2 percent of the applied pressure increment. This is shown graphically in Figure 2(a). A test to determine the altimeter and airspeed indicator lag constants can be accomplished on the ground by applying a suction sufficient to develop a ΔH_p equal to 500 ft or an indicated airspeed of 100 kt. Removal of the suction and timing the altitude drop to 184 ft, or the airspeed drop to 37 kt, results in the determination of λ_s, the static pressure lag constant, as shown in Figures 2(b) and 2(c). If a positive pressure is applied to the total pressure pickup (drain holes closed) to produce a 100 kt indication, the total pressure system lag constant (λ_T) can be determined by measuring the time that is required for the airspeed indicator to drop to 37 kt when the pressure is removed. Generally, the λ_T will be much smaller than the λ_S because of the small volume of the airspeed instrument case.

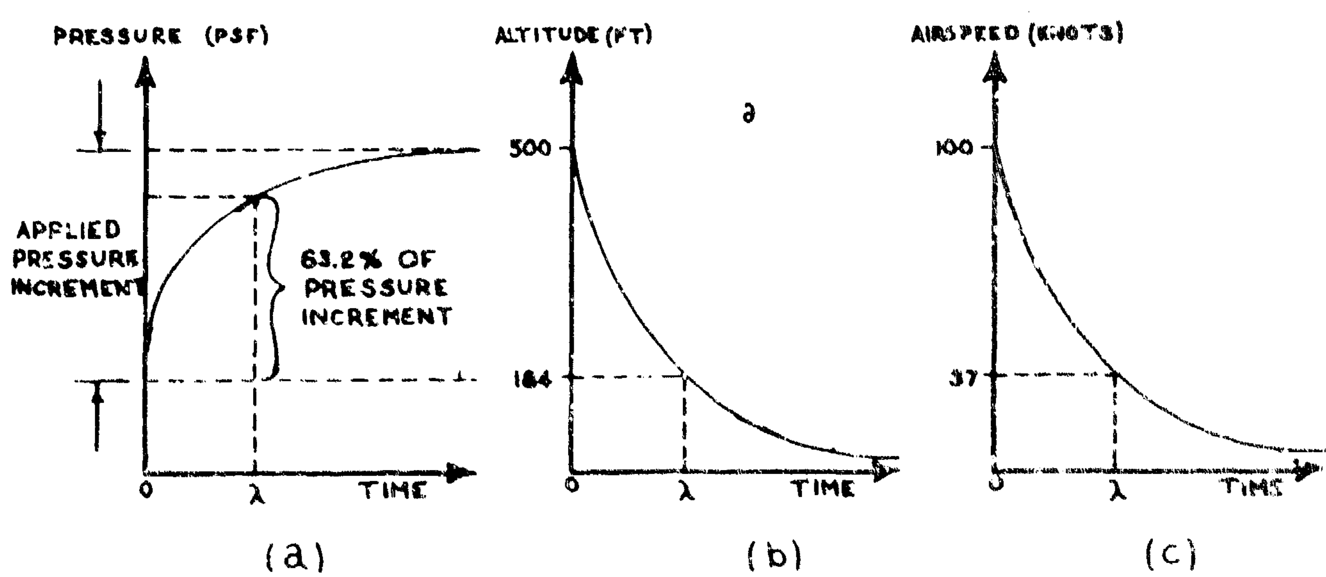


Figure 2
Determination of Pitot/Static System Lag Constant (λ)

System Balancing

The practical approach to lag error testing is to determine if a serious lag error does exist and to eliminate it where possible. To test for airspeed system balance, a small increment of pressure (0.5 psf) is applied simultaneously to both the pitot and static systems. If the airspeed indicator does not fluctuate, the combined systems are balanced and no lag error will exist in indicated airspeed data because the lag constants are matched. Movement of the airspeed pointer indicates that the addition of more volume is required in one of the systems. The addition of cans or tubing (see Figure 1) will generally provide satisfactory airspeed indications but will not help the lag in the altitude indicator as this lag is primarily caused by the length of the static system tubing. For instrumentation purposes, lag can be eliminated from the altimeter by locating a static pressure recorder remotely at the static port.

Position Errors

Position errors or "installation errors" are the result of other than free stream pressures at the pressure sensor or errors in the local pressure at the source resulting from the shape, location, or orientation of the sensor. In an airplane these pressure errors are usually present only at the static pressure sensor and the total pressure measurement (pitot tube) is assumed to be error-free. The limits for this assumption are discussed in NACA Report 919.

Based on the assumption of an error-free total pressure measurement, airspeed and altimeter systems may be calibrated by either the altimeter depression method or by the pace method. Since the altimeter depression methods are generally more economical in terms of flight test time and do not require a calibrated pace

airplane, they are most often used. Differences between the pressure measured at the static source and the true ambient pressure result in an early determined static source position error. The effects on airspeed of errors in static pressure measurement may be best understood by examining the following equations for calibrated and indicated airspeed.

$$V_c = \left\{ \frac{2\gamma P_{SSL}}{(\gamma - 1)\rho_{SSL}} \left[\left(\frac{P_T - P_A}{P_{SSL}} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \right\}^{1/2}$$

$$V_i = \left\{ \frac{2\gamma P_{SSL}}{(\gamma - 1)\rho_{SSL}} \left[\left(\frac{P_T - P_S}{P_{SSL}} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \right\}^{1/2}$$

These equations show that a static pressure (P_s) value that is lower than the true ambient pressure (P_a) will result in an indicated airspeed (V_i) that is greater than the calibrated airspeed (V_c). If the static pressure is higher than ambient pressure, the reverse is true. In each case, the total pressure (P_T) measurement is assumed to be error-free.

Other altimeter depression methods which may be used are the tower fly-by, the use of the trailing cone or trailing bomb, the stadiametric camera fly-by, and the use of the radar altimeter. The fly-by method is typical of the altimeter depression methods and is relatively simple to perform and analyze. It is the method now being used at TPS.

The theodolite fly-by test consists of a series of stabilized level flight runs at several airspeeds over a fixed course. For the Point-No-Point range, theodolite film readings of the actual altitude (h_{tape}) of the test airplane for each run are obtained using a contraves phototheodolite. For the Webster Field range, actual airplane

altitude is determined using a special camera which produces polaroid pictures with a grid superimposed over the image. The altimeter position error is determined by comparing the actual pressure altitude of the test airplane (tapeline altitude corrected for temperature variations from standard) with the indicated pressure altitude of the test airplane. The airspeed position error (caused by the static source error) may be calculated directly from the altimeter position error with the assumption of no total pressue error. For airspeeds less than approximately 0.6M, the airspeed position error is mainly a function of angle of attack, and the static source position error coefficient ($\frac{\Delta P}{q_{c_i}}$) determined in this test will be valid for all altitudes at standard gross weight. From the static source position error coefficient determined for the standard weight, the airspeed position error and the altimeter error may be calculated for other gross weights. For constant indicated airspeed, hence constant angle of attack, the altimeter position error increases with increasing altitude. This altimeter position error variation can be calculated and altimeter position error corrections for all altitudes and weights can be extrapolated from the data (static source position error coefficients) obtained at one altitude.

Extrapolation of airspeed and altimeter position error corrections based on sea level data may not yield acceptable results in all cases. Extrapolation requires certain simplifying assumptions, e.g., negligible total pressure source error and Reynold's number effects. For large extrapolations (sea level to 40,000 ft) these assumptions may fail, and the position error correction for a turbojet airplane determined from sea level data obtained must be verified by additional tests at high altitude using a calibrated pacer airplane.

Military Specifications

Military Specification MIL-I-6115A, Amendment 3, of 31 December 1960 states the allowable limits of the altimeter and airspeed position error:

a. Airspeed error - ±4 kt at all airspeeds.

b. Altitude error - 25 ft per 100 KIAS.

DOD Joint Chief of Staff memo of 2 July 1963 stipulates that for all military aircraft "THE MODE C ALTITUDE INFORMATION SHALL BE OBTAINED FROM AN ALTIMETER WITH A MAXIMUM ALTIMETER SYSTEM ERROR OF PLUS OR MINUS 250 FEET AT ALL SPEEDS AND OPERATING ALTITUDES WHILE IN THE MISSION OF FLIGHTS. ALTIMETER ACCURACIES IN OTHER PHASES OF FLIGHT, i.e., LANDING, TAKEOFF, INSTRUMENT APPROACHES, etc., SHALL MEET PRESENT MILITARY/FAA TSO SPECIFICATION STANDARDS."

Probe Recovery Factor

The OAT is a necessary flight test parameter and is used in conjunction with airspeed and altitude to determine true airspeed. The OAT probe recovery factor can be determined while conducting airspeed and altimeter position error tests. The indication of the OAT gauge is a function of the ambient air temperature, the recovery factor of the OAT system's probe, and the airplane Mach number. The tower fly-by test is flown over a range of Mach numbers wide enough to determine the recovery factor (K) from the OAT readings observed during the test. Since the recovery factor is a measure of the efficiency of the probe design in preventing heat loss by radiation, it should become obvious that a recovery factor greater than 1.0 cannot be obtained. Typical recovery factors range between 0.85 to 0.98. Recovery factors for special flight test instrumentation should be at least 0.95.

Data Recording

a. This test requires three observers in the ground station. Two observers are required to operate the camera equipment while a third observer calls "MARK" when the test airplane is over the Point-No-Point lighthouse or abeam-the-Webester-Tower. Technical Support Division (TSD) will supply the camera operators. TPS is required to supply the third observer.

b. Photopanel data are generally more accurate than cockpit observed data and will be used when available.

c. Sequence of Test Data: In configuration CR, commence runs at a mid-range airspeed and work up to maximum airspeed. Then work down from the initial airspeed to the slowest airspeed desired or allowed. For configuration PA, begin at the maximum limit airspeed and work down to minimum airspeed. Correlate runs with other airplanes in the pattern to optimize safety and utilization of the ranges.

Preflight Procedures

a. Required Data:

1. Pilot - Kneeboard data card (Figure 3).

V_o, H_{p_o} at "MARK" called by the range observer.

OAT_o (record for configuration CR points only).

Configuration.

Fuel remaining or fuel used.

Photopanel counter number.

Photopanel (if available).

Energize 10 sec prior to data point.

Turn OFF while climbing downwind.

2. Range Observer (prepare data card for each airplane).

Run No.

Theodolite end frame number for each run.

Time.

Ambient air temperature at beginning and end of tower runs (information only).

b. Know the following:

Airplane configuration details.

Airspeed ranges for each configuration.

Airplane restrictions.

Traffic pattern.

c. Prepare data card (see Figure 3).

d. Inspect photopanel for loading and operation.

e. Note location of pitot tube and static sources in order to reference the applicable Military Specification and to discuss the test results properly.

WEATHER

Sample Data Card

CARD NO.

Airspeed Calibration

AIRPLANE TYPE: UPF-7

BU. NO.: 117711

TIME T.O.

TIME LAND

DATE: 9 Oct 78

GEAR DOWN UP

PILOT: Delong

CONDITION: Normal Fighter

T.O. GROSS WEIGHT

EXTERNAL CONFIGURATION

Run	Config	V_{tgt}	V_o	H_{po}	OAT_o	Fuel	CTN	Remarks
1	CR	175						
2	↓	200						
3	↓	225						
4	↓	250						
5	↓	275						
6	↓	V_{MAX}						
7	↓	150						
8	↓	125						
9	↓	100						
10	PA	125						
11	↓	115						
12	↓	105						
13	↓	95						
14	↓	90						
15	↓	85						
16	↓	80						

FLIGHT DATA PNHC NATC 1900 3

Figure 3
Sample Data Cards

Flight Procedures for Point-No-Point Range

a. Inform ground control that you will be operating in R4005, 1,500 feet to the surface, in the vicinity of POINT LOOKOUT. Closest point of approach to the field will be 7 miles.

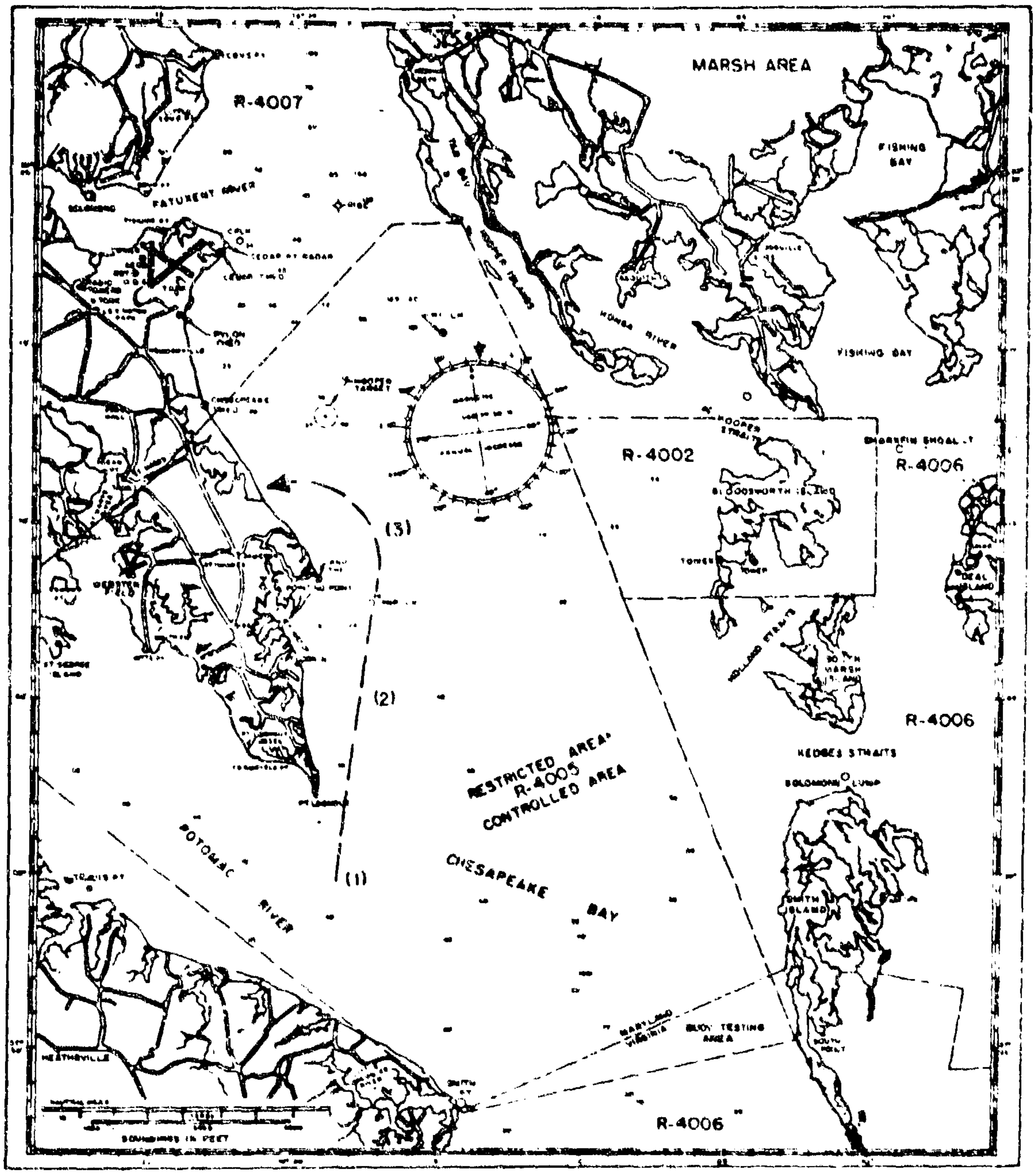
b. After takeoff obtain clearance from Approach Control (354.8/281.8 mcs) for clearance into R4005. Approach Control will shift you to project frequency (385.2/380.4). Contact TANGO control (Point-No-Point Tower) on assigned frequency and enter pattern on a 1,000-1,500 ft downwind leg (see Figure 4). Fly the pattern depicted, descending to target altitude (500 ft) when established on the inbound track. Call SIDE NUMBER, RUN NUMBER, and TARGET AIRSPEED when inbound near POINT LOOKOUT. Call a 10 SEC STANDBY prior to passing over the lighthouse. The tower observer should call a "MARK" when your airplane is directly over the lighthouse and will record your run number for correlation of data.

c. Make the runs at approximately the target altitude in airspeed increments to cover the range of airspeeds to be investigated in the time allowed. To obtain an accurate measurement of height of the airplane above the lighthouse, it is essential that the flight path be along a straight track over the lighthouse. The target altitude for each run should be 500 ft AGL. For each run, allow sufficient straight-away to ensure that airspeed and altitude are stabilized 10 sec before passing over the lighthouse. Stabilized flight and good trim are essential to obtain satisfactory data in this test.

d. After passing the lighthouse, be alert for traffic. Make a climbing left turn and maintain 1,000-1,500 ft AGL downwind. Do not descend below 1,000 ft until clear of land.

e. For each run record the observed altitude (H_{p_o}) and observed airspeed (V_o) as the airplane passes over the lighthouse. When the photopanel is used, it should be energized about 10 sec before passing the lighthouse and switched OFF when over the lighthouse. This procedure results in a record of degree of stabilization in the final approach with the last frame of this film burst being the actual data used. Time and OAT may be recorded as soon as airspeed has stabilized on the inbound leg. Record film counter number, configuration, and fuel on the downwind leg.

f. Photopanel data is desirable for this test; however, flight reports will be submitted on kneepad data in case of photopanel malfunction.



WST RANGE—CONTROLLED AREA

Notes:
(1) Obtain clearance from NAS Patuxent Approach Control
(2) Airspeed and altitude stabilized 10 sec before crossing PNP lighthouse. Inbound heading approximately 010° magnetic or as desired by the tower.
(3) Climbing turn downwind to at least 1,000 ft. CAUTION-conflicting traffic.

Figure 4
Point-No-Point Theodolite Course

Flight Procedures for Webster Field Range

a. Inform ground control that you will be operating in R4005, 1,500 ft to the surface, in the vicinity of WEBSTER FIELD. Closest point of approach to Patuxent will be 7 miles.

b. After takeoff, obtain clearance from Approach Control (354.8/281.8 mcs) for clearance into R4005. Approach Control will shift you to Webster Tower Frequency (358.0). Enter pattern on a 1,000-1,500 ft downwind leg. Fly the pattern so as to remain above 1,000 ft over land, altitude not less than 150 ft AGL when established on the inbound track (see Figure 5). Call SIDE NUMBER, RUN NUMBER, and TARGET AIRSPEED when inbound abeam St. Georges Island. Call a 10 SEC STANDBY prior to passing Webster Tower. The tower observer should call a "MARK" when your airplane is directly abeam the tower and will record your run number for correlation of data.

c. The target altitude for each run should be 200 ft AGL. Make the runs in airspeed increments to cover the range of airspeeds to be investigated in the time allowed. For each run, allow sufficient straight-away to ensure that airspeed and altitude are stabilized 10 sec before passing abeam the tower. Stabilized flight and good trim are essential to obtain satisfactory data in this test.

d. After passing the Tower be alert for traffic. Make a climbing left turn and maintain 1,000-1,500 ft AGL downwind. Do not descend below 1,000 ft until clear of land.

e. For each run, record the observed altitude (H_{p_o}) and observed airspeed (V_o) as "MARK" is called. When the photopanel is used, it should be energized about 10 sec before passing abeam the tower and switched OFF when "MARK" is called. This procedure results in a record of degree of stabilization in the final approach

with the last frame of this film burst being the actual data used. Time and OAT may be recorded as soon as airspeed has stabilized on the inbound leg. Record film counter number, configuration, and fuel on the downwind leg.

f. Photopanel data are desirable for this test; however, flight reports will be submitted on kneepad data in case of photopanel malfunction.

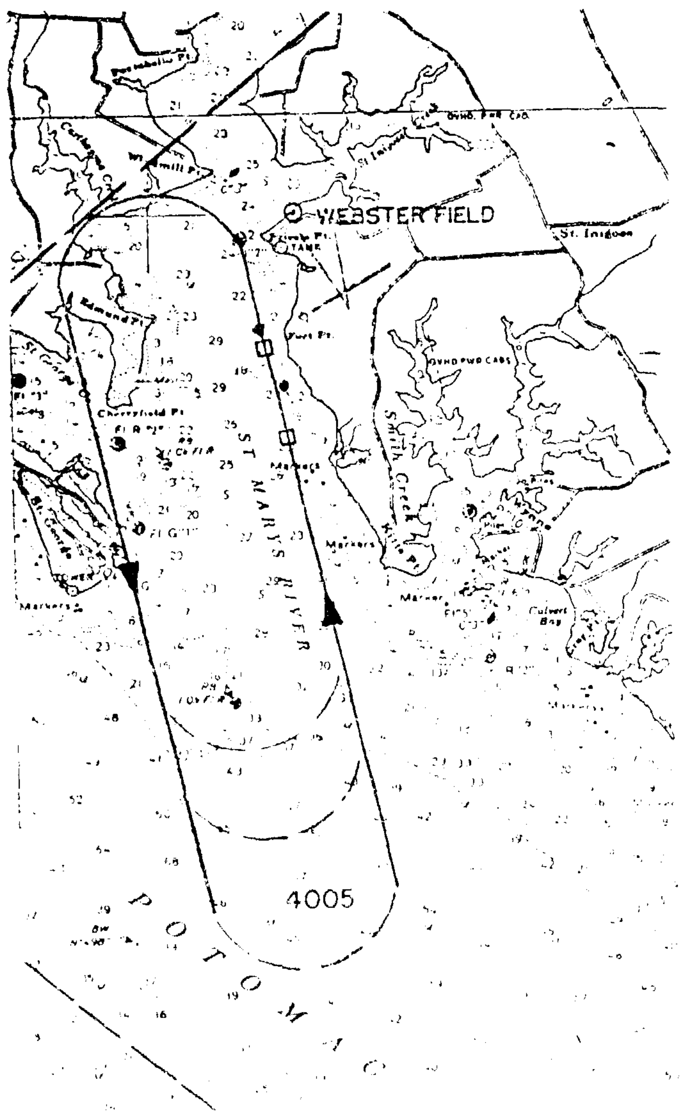


Figure 5
Webster Field Range

Tower Observer Procedures

a. Ensure a sensitive altimeter is available.

b. Arrange own transportation to the ground stations. Dress warmly in cold weather -- both towers are well ventilated.

c. Monitor control frequency. Make transmissions as required for coordination, call "MARK" as each data point is taken, and aid in recording data. Record the pressure altitude of the theodolite ($H_{p_{theo}}$) for each run, using the sensitive altimeter in the tower. Obtain ambient air temperature (shaded area).

Postflight Procedures

a. Run photopanel film out after landing.

DATA REDUCTION AND PRESENTATION

General

Airspeed and altimeter position error corrections for the standard gross weight will be determined for both configurations tested. The OAT probe recovery factor will also be determined. The first and last data points must be worked by hand.

Point-No-Point Procedures

From the theodolite observer's data sheet record the ambient temperature in the tower. From the theodolite film (see Figure 6) determine:

a. Frame counter number.

b. The elevation angle of the cross hairs.

c. The error in tracking the airplane. This error in feet may be determined directly from the film by using the airplane image to scale the film. When the cross hairs are below the airplane, the error is positive.

From the theodolite observer's data sheet and the information recorded from the theodolite film calculate:

a. Theodolite cross hair tapeline height above the tower; use the cross hair elevation angle and table 1.

b. Airplane altimeter tapeline height above the tower; cross hair height plus tracking error (Figure 7).

c. T_a ($^\circ$F); use OAT data obtained at a low Mach number to approximate this value.

d. $H_{p(tower)} = H_{p_o} + \Delta H_{p_{ic}}$

e. Airplane pressure height above mean sea level

$$h_{pc_{airplane}} = h_{p_{tower}} + \frac{T_{std}}{T_{test}} \left[\overbrace{h_{tapeline_{airplane}} - h_{tapeline_{tower}}}^{\Delta h_{tape\ above\ tower}} \right]$$

Where T_{std} is the standard day temperature at the tower (T_{std} and T_{test} are in deg Rankine).

From kneeboard or photopanel data record the following information:

a. Altitude: H_{po}

b. Airspeed: V_o

c. Fuel used or Fuel remaining

d. Outside air temperature: OAT_o

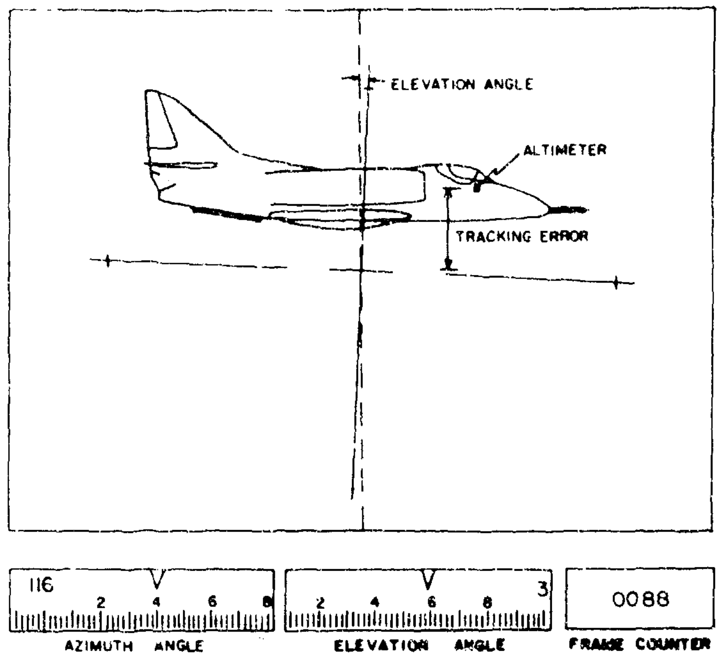


Figure 6
Point-No-Point Theodolite Film Data

Table 1

Theodolite Elevation Angle vs. Tapeline Altitude Above

θ ELEV. ANGLE (DEG)	TAPE-LINE ALT (FT)	θ ELEV. ANGLE (DEG)	TAPE-LINE ALT (FT)	θ ELEV. ANGLE (DEG)	TAPE-LINE ALT (FT)	θ ELEV ANGLE (DEG)	TAPE-LINE ALT (FT)	θ ELEV. ANGLE (DEG)	TAPE-LINE ALT (FT)
0	0	1.0	179	2.0	359	3.0	538	4.0	718
.1	18	1.1	197	2.1	376	3.1	556	4.1	736
.2	36	1.2	215	2.2	394	3.2	574	4.2	754
.3	54	1.3	233	2.3	412	3.3	592	4.3	772
.4	72	1.4	251	2.4	430	3.4	610	4.4	790
.5	89	1.5	269	2.5	448	3.5	628	4.5	808
.6	107	1.6	287	2.6	466	3.6	646	4.6	826
.7	125	1.7	305	2.7	484	3.7	664	4.7	844
.8	143	1.8	323	2.8	502	3.8	682	4.8	862
.9	161	1.9	341	2.9	520	3.9	700	4.9	880
1.0	179	2.0	359	3.0	538	4.0	718	5.0	898

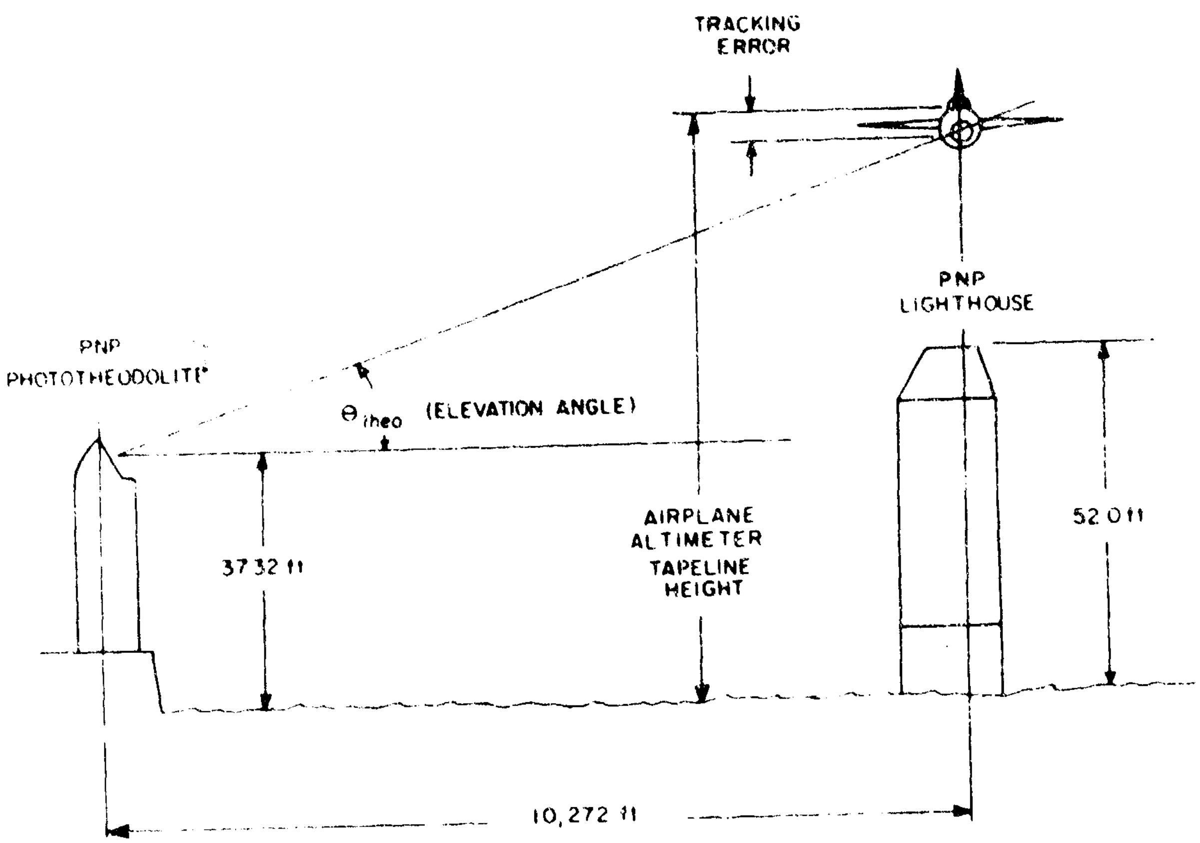


Figure 7
Point-No-Point Theodolite Survey Data

Webster Field Procedures

a. From the tower observer's data sheet, record the ambient temperature (^{o}F) outside the tower.

b. From the tower polaroid film determine the scale length (inches) of the test airplane and the scale height (inches) of the test airplane above the reference line (Figure 8).

c. Determine the actual length (ft) of the test airplane from Appendix I. (Airplane Details)

d. Calculate the tapeline height (ft) of the test airplane above the reference line by the expression.

$$h_{\substack{\text{tape}\\ \text{above reference}}} = \left[\frac{\text{airplane length (ft)}}{\text{scale length (inches)}}\right] \text{scale height}_{\substack{\text{above reference}\\ \text{(inches)}}}$$

e. Calculate the pressure altitude in the tower by

$$h_{p_{tower}} = h_{p_o} + \Delta h_{p_{ic}}$$

f. Calculate the calibrated pressure altitude of the test airplane by the expression

$$h_{pc_{airplane}} = h_{p_{tower}} + \frac{T_{std}}{T_{test}} \left[\overbrace{h_{\substack{\text{tapeline}\\ \text{airplane}}} - h_{\substack{\text{tapeline}\\ \text{tower}}}}^{\Delta h_{\text{tape above tower}}} \right]$$

where

T_{std} is the standard day temperature at the tower.

T_{std} and T_{test} are in deg Rankine.

<u>Position Error Determination</u>

a. $H_{p_i} = H_{p_o} + \Delta H_{p_{ic}}$

b. $V_i = V_o + \Delta V_{ic}$

c. $P_i = f(H_{p_i})$ (Use tables of P vs. H_p in Appendix IX.)

d. $P_A = f(H_{p_c})$ (Use tables of P vs. H_p in Appendix IX.)

e. $\Delta P = P_S - P_A$

f. $q_{c_i} = f(V_i)$ (Use charts of q_c vs. V_c in Flight Test Reference Book.)

g. Calculate $\dfrac{\Delta P}{q_{c_i}}$

h. $W_t = W_{FULL} - \left[\text{Fuel used}\right]$ or $\left[W_{EMPTY} + \text{Fuel remaining}\right]$

i. $\dfrac{W_s}{W_t}$ (Use W_s from Airplane Details in Appendix I.)

j. $V_{i_w} = V_i \left(\dfrac{W_s}{W_t}\right)^{1/2}$

Plot the variation of P/q_{c_i} with V_{i_w} and fair a smooth curve through the data. Label this curve for the standard weight.

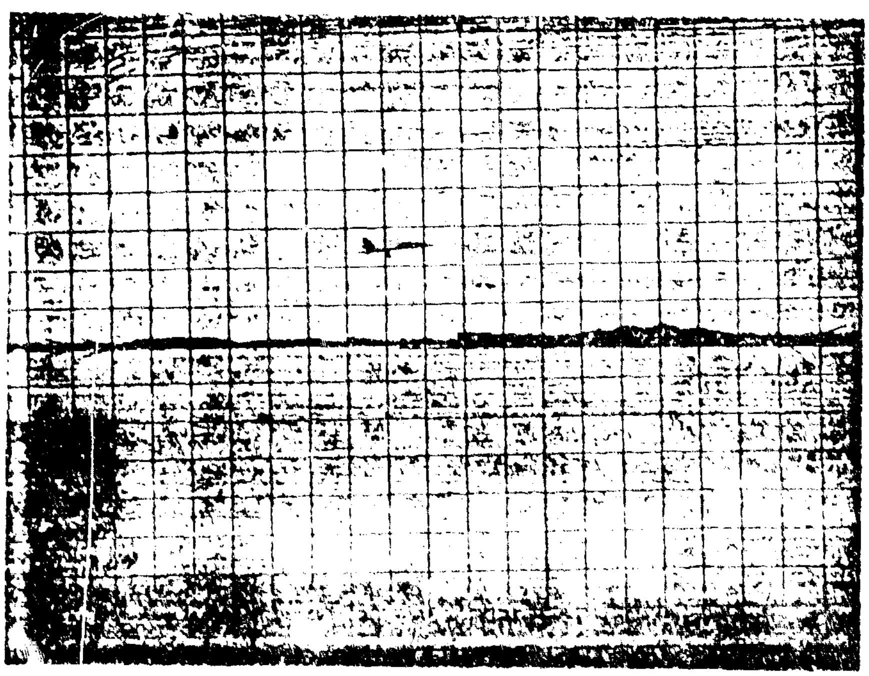


Figure 8
Polaroid Film Data from Webster Field

The military specification requires that position error be determined at sea level (standard conditions) at a standard gross weight. In order to determine specification compliance, use the faired curve of $\Delta P/q_{c_i}$ vs. V_{i_w} to determine ΔP for various V_{i_w} $\left[\Delta P = P/q_{c_i} \text{ where } q_{c_i} = f(V_{i_w}) \text{ since } V_{i_{test}} = V_{i_w} \right]$. Determine and plot the variation of Δpos with V_{i_w} where $\Delta pos = V_c$ (based on $q_{c_i} + \Delta P$) less V_i (in this case equal to V_{i_w}). Also determine and plot for sea level only the variation of $\Delta H_{pos} = H_{pc}$ less H_{p_i} (based on $P_a + \Delta P$).

<u>Determination of OAT System Recovery Factor</u>

a. Calculate q_c for each data point flown

$$q_c = q_{c_i} + \Delta P$$

b. $V_c = f(q_c)$ (Use charts of q_c vs V_c in Flight Test Reference Book.)

c. Determine Mach number from V_c, H_p, and M. Use V_c from step b and H_{p_c} determined earlier.

d. Calculate M^2

e. Correct OAT_o for instrument error

$$OAT_i = OAT_o + OAT_{ic}$$

f. Plot OAT_i against M^2 and draw a straight line through the data extending the line so that it intersects the line of zero M^2.

g. This plot is a graphical solution of the OAT equation:

$$OAT = T_a \, (1 + 0.2 \, KM^2) - 273, \qquad °C$$

h. Measure the slope, m, of the straight line.

i. Calculate T_a from t_a determined in step e

$$T_a = t_a + 273, \ ^\circ K$$

j. Determine recovery factor, K

$$K = \frac{m}{0.2T_a}$$

Presentation of Results

The following curves should be presented. Sample presentations are included in the Guidelines to Flight Reports, and a sample data reduction sheet is included as Figure 9.

a. $\Delta H_{P_{pos}}$ versus V_i

b. ΔV_{pos} versus V_i

c. $\frac{\Delta P}{q_{c_i}}$ versus Vi

d. M^2 versus OAT_i

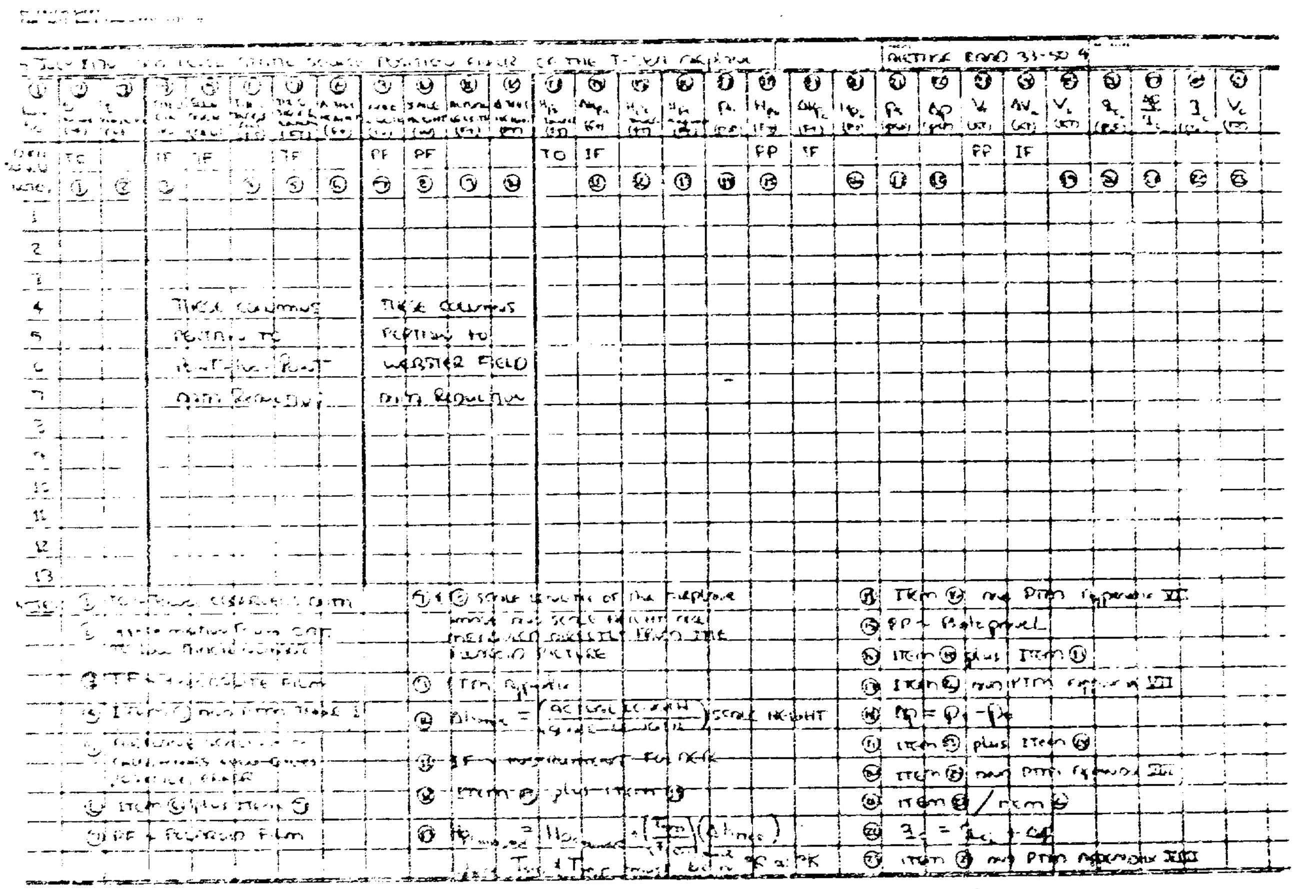


Figure 9
Sample Data Reduction Sheet

[illegible] SEA LEVEL STATIC SOURCE POSITION ERROR ON THE T-37A AIRPLANE | | ARTISK RMD 33-50-4

(1) Test No.	(2) W	(3) V_{ind} (kt)	(4) V_{cal} (kt)	(5)	(6) V_{ic} (kt)	(7) OAT (°C)	(8) ΔT (°C)	(9) OAT_c (°C)	(10) H_{pi} (ft)	(11) V_{ic} (kt)	(12) q_{c_i} (psf)	(13) $\frac{\Delta P}{q_{c_i}}$	(14) Δp (psf)	(15) P_o (psf)	(16) P_s (psf)	(17) H_{pc} (ft)	(18) ΔH_{pc} (ft)	(19) q_c (psf)	(20) V_c (kt)	(21) ΔV_{pc} (kt)
Data Source		FP				FP	TF		AD	AC										
Notes	(a)	(b)	(c)		(d)			(e)	(f)	(g)	(h)	(i)	(j)	(k)	(l)	(m)	(n)	(o)	(p)	(q)
1																				
2																				
3																				
4									→ THESE COLUMNS PERTAIN											
5									TO SPECIFICATION COMPLIANCE											
6									→ AT SEA LEVEL (STANDARD											
7									CONDITIONS)											
8									→											
9																				
10																				
11																				
12																				
13																				

NOTES:

(a) Item (3) and item (5) from PTM APP III

(b) Fill in either fuel used or fuel remaining

(c) W_{act} = W_{ramp} − Fuel used or W_{act} = W_{empty} + Fuel Remaining

(d) V_{ic} = $V_{c\,std}$ $(W_{std}/W_{act})^{1/2}$ where W_{std} is obtained from PTM

(e) Item (8) plus item (7)

(f) AD: Assumed Data

(g) The results of the specification requirements are stated for standard sea level conditions at the aircraft standard weight. Here H_{pc} is assumed 0.

(h) Assumed points (e.g. 150, 200, 250, ...)

(i) Item (11) from PTM App VIII

(j) From faired curve of $\Delta P/q_{c_i}$ vs V_{ic}

(k) Item (12) × item (13)

(l) Item (10) from PTM App VII

(m) P_s = P_o + Δp

(n) Item (16) from PTM App VII

(o) ΔH_{pc} = H_{pc} − H_i

(p) q_c = q_{c_i} + Δp

(q) Item (19) from PTM App VIII

(r) ΔV_{pc} = V_c − V_{ic}

SECTION V

STALL AIRSPEED DETERMINATION

REFERENCES

Section V

1. USNTPS - FTM - No. 102, Section II.

2. Flight Test Technical Memorandum 72-1, "Determination of Performance Stall Airspeed at Altitude and Sea Level."

STALL AIRSPEED DETERMINATION BY THE PACING METHOD

PURPOSE

The purpose of this test is to determine the stalling airspeed of the test airplane in the takeoff and landing configuration. The variation of indicated stalling speed with airplane gross weight is required for presentation in the flight handbook. The calibrated stall airspeed will be used to determine compliance with the contract guarantee.

DISCUSSION AND THEORY

In the classic aerodynamic one g stall, the coefficient of lift increases as the airspeed decreases until an angle of attack is attained at which the lift abruptly diminishes and the airplane ceases to fly. The stall airspeed for this case is well defined. In many airplanes, the classic stall does not occur. In these cases, the airplane will exhibit some characteristic at low airspeed which defines a minimum useable flying speed. This characteristic may be loss of effective control about one or more axis or a high sink rate. Although the aerodynamic stall has not occurred, the minimum useable flying speed is, in effect, the contract stalling speed of the airplane. Frequently, this airspeed is not well defined and is subject to interpretation by the particular pilot concerned. Therefore, it is most important that the stall speed for a particular airplane be defined consistently.

A pilot is primarily concerned with the indicated airspeed at stall, and this value is easily determined from the indication of the service airspeed system. In performance testing, accurate calibrated stall airspeeds are required to determine liftoff and obstacle clearance airspeeds for takeoff performance and are used in

determining carrier launch criteria. In addition, they are used as an incentive or penalty for airplane contractors (performance guarantees). Because of the large angles of attack attendant to the stall, the service airspeed system position error usually is large and not accurately known. One means of determining the stall airspeed precisely is by equipping the test airplane with a special pitot-static system with a known position error. Another method is to measure the stall airspeed of the test airplane from a pacer airplane which has a lower stall speed. The pacer airplane is equipped with a special pitot-static system for which the position error is accurately known.

In an airplane loaded at one center of gravity (cg) location, the stall will occur at the same coefficient of lift regardless of the airplane gross weight. Unless large thrust effects are present, the stalling airspeed will vary with the square root of the airplane gross weight. This relationship can be used to extrapolate stall speed data for one gross weight to other gross weights. If the cg location moves as the airplane weight changes, the coefficient of lift at stall also will change and the square root relationship will be approximate only. Another factor which influences stall speed is altitude. As altitude increases, the maximum lift coefficient decreases because of Reynold's number and Mach number effects. This results in an increase in stall speed as altitude increases; however, this effect is usually negligible for altitudes below 10,000 ft.

Nonsteady aerodynamic flow effects must be considered in the determination of stall airspeed. These effects are caused by the change in vorticity over the actual lifting surface. Any change in vorticity must take place over some finite period of time. The problem has been treated by a number of researchers, and analysis shows that airfoils changing flow patterns in distances less than 25 chord

lengths experience appreciable nonsteady flow effects. A satisfactory first approximation for calculating the ratio of actual to steady state lift values following a step change in airspeed is shown in equations (1) and (2). Equation (1) is for accelerations and equation (2) is for decelerations.

$$\frac{L}{L_o} = \frac{R + 1}{R + 2} \tag{1}$$

$$\frac{L}{L_o} = \frac{R + 2}{R + 1} \tag{2}$$

L is lift after travel of R semi chord lengths, and L_o is steady state lift after travel of an infinite number of semi chord lengths. For the decelerating case, this chord travel can be expressed in terms of airspeed as shown in equation (3).

$$V_S = \sqrt{\frac{R + 1}{R + 2}} \, (V_{o_s}) \tag{3}$$

For uniform accelerations, the relation of actual to steady state lift is a more complex relationship, but still retains a primary relationship to chord lengths traveled. The acceleration case has received considerable attention with regard to catapult design. The deceleration case has been generally neglected in technical literature. An empirical approach for determining R that appears to work well is proposed in equation (4).

$$R = \frac{V_S}{\frac{c}{2}\dot{V}} \tag{4}$$

Here, V_s is the measured (test) stall airspeed in KIAS; $\dot{V}$ is deceleration rate in kt/sec, and c is the chord in feet.

An example which demonstrates both the use of these equations and the importance of a small deceleration rate follows.

$$V_s = 114 \text{ KIAS}$$

$$V = -0.5 \text{ kt/sec}$$

$$\frac{c}{2} = 3 \text{ ft}$$

therefore,

$$R = \frac{114}{3(0.5)} = 76$$

$$V_{O_S} = \frac{76 + 2}{76 + 1} (114) = 114.4 \text{ KIAS}$$

However, at a deceleration rate of 5 kt/sec, the test stall airspeed is 108 KIAS. In this case

$$R = \frac{108}{3(5)} = 7.2$$

$$V_{O_S} = \frac{9.2}{8.2} (108) = 114.4 \text{ KIAS}$$

The example shows the importance of a deceleration rate that is no greater than 1 kt/sec, and that in the airspeed range of these test results, a 5 kt/sec deceleration yields a test stall airspeed that is 6.4 kt too low. Such results please contractors, but are of little test value.

TEST PROCEDURES AND TECHNIQUES

Pertinent Particulars

a. Approach stall speed at a deceleration rate of 1 kt/sec or less (1/2 kt/sec preferred).

b. Alternate the configurations being tested.

c. All data will be kneeboard data (no photopanel).

d. Self-paced flights may be conducted in appropriately instrumented airplanes capable of obtaining calibrated stall airspeeds. Attempt to achieve the maximum possible difference between the gross weight at the heaviest data point and the gross weight at the lightest data point.

Preflight Procedures

a. Prepare pilot's and pacer observer's data cards. In flight, at each stall, record the following data:

Test Airplane	Pacer
Run No.	Run No.
Airspeed	Airspeed as read from special airspeed system
Fuel used or remaining	Pacer configuration
Configuration	Remarks as to validity of data point (i.e., any relative motion between airplanes)
Pressure altitude	

b. Brief with pacer pilot as to rendezvous point and altitude, radio frequency, formation position, and hand signals in case of communication failure.

c. Review stall approach technique and stall recovery procedures.

Flight Procedures

Before the pacer joins the test airplane, the test pilot should perform at least one stall in each configuration to observe stall characteristics. If the airplane tends to pitch up or roll at the stall, instruct the pacer to join on the "safe" side. When cleared by the test airplane pilot, the pacer should take an echelon formation position, but not so wide that quick detection of relative motion is lost.

When the pacer is in position, the thrust of the test airplane should be smoothly set for the desired configuration and a steady deceleration of 1/2 kt/sec or less commenced. Control the deceleration rate by adjusting rate of climb or descent. A few seconds before stall call out "STAND-BY" and at the stall "MARK." Recover from the stall and record data.

As the test airplane decelerates, the pacer must maintain a constant relative position. For the landing configurations, the pacer will usually require landing gear and flaps DOWN and speed brakes EXTENDED in order to maintain position. If the pacer tends to overrun the test airplane, the best results are obtained by allowing the test airplane to descend below the pacer while maintaining a constant position in the vertical plane. If there should be relative motion at the stall, make an estimate of the relative speed and record this information on the observer's data card. Keep the test airplane in sight during the stall recovery, but do not dive to follow him. A quicker rendezvous is accomplished when the pacer maintains his altitude. When the pacer is again in position, repeat the test in the other configuration. Continue to conduct paced stalls in alternate configurations until the required data is obtained or minimum fuel is reached.

DATA REDUCTION

General

Report the variation of calibrated and indicated airspeed at stall with gross weight for the test airplane. Determine compliance with the stall speed guarantee. Record the following data on the data reduction sheet:

V_{o_t} V_{o_p} H_{p_o} Configuration Remarks

Data Reduction and Presentation

a. Correct pacer observed airspeed for instrument and position error to determine calibrated airspeed at the stall.

b. Correct test airplane observed airspeed to instrument error to determine indicated airspeed at the stall.

c. Calculate test airplane gross weights for each stall.

d. Plot the variation of indicated airspeed with gross weight for configurations Takeoff and Landing. Plot the variation of calibrated airspeed with gross weight for configuration Landing and indicate the contract guarantee.

SECTION VI

DETERMINATION OF EXCESS POWER CHARACTERISTICS

SECTION VI

NOTATIONS INTRODUCED IN THIS SECTION

ΔHp	Difference in pressure altitude at the upper and lower limit of an altitude band centered at the test altitude
R/C_i	Average indicated rate of climb, $\Delta H_{pi}/\Delta t$
W_A	Average test gross weight, $\frac{\text{sum of test weights at each point}}{\text{number of points}}$
E_h	Energy height, or specific energy
dE_h/dt	Rate of change of energy height
h	Geometric altitude, ft
H_p	Pressure altitude, ft
s	Subscript signifying standard day conditions and standard airplane gross weight
Δt	Increment of time required to climb through test altitude band
P_s	Specific excess power, equals dE_h/dt

REFERENCES

SECTION VI

1. AGARD Vol. I, pp. 4:37-4:40 and 7:27-7:31

2. Dommasch, Airplane Aerodynamics, pp. 289-290

3. Rutowski, DAC, "Energy Approach to the General Aircraft Performance Problem," 1953

4. Boyd, J. R., MAJ, USAF, "Energy - Maneuverability," 1966

DETERMINATION OF EXCESS POWER - LEVEL ACCELERATION RUNS AND SAWTOOTH CLIMBS

PURPOSE

The purpose of conducting acceleration runs and sawtooth climbs is to determine the variation of excess power with airspeed. For acceleration runs, the data will be used to evaluate level acceleration characteristics and maximum level flight airspeed, to determine one or more climb schedules, and can be used to estimate level flight turning performance. For sawtooth climbs, the data will be used to evaluate climb characteristics.

DISCUSSION AND THEORY

General

In analyzing acceleration or climb performance, it is advisable to consider the airplane from the standpoint of total energy (TE) which is the sum of potential (PE) and kinetic (KE) energies. The total energy of an airplane at a given weight (W), airspeed (V), and geometric altitude *(h) is,

$$TE = PE + KE = Wh + \frac{W}{2g} V^2 \qquad (1)$$

dividing by W and defining energy height as:

$$E_h = E/W = h + \frac{V^2}{2g} \qquad (2)$$

*Geometric altitude assumed equal to geopotential altitude

Energy height, or specific energy, can also be interpreted as energy altitude; the altitude which could be attained if all the kinetic energy were perfectly converted to potential energy. Differentiating, we obtain:

$$\frac{dE_h}{dt} = \frac{dh}{dt} + \frac{V}{g}\frac{dV}{dt} \qquad (3)$$

Now consider the free-body diagram of an airplane in climbing flight where the thrust (F) is assumed to be along the flight path:

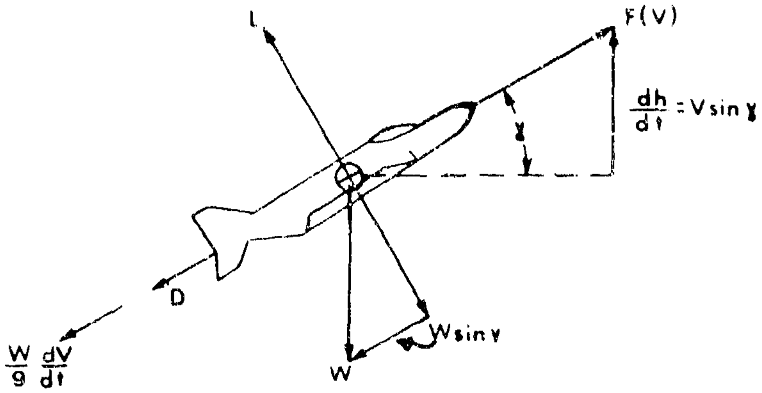


Figure 1
Free-Body Diagram

$$\Sigma F = \frac{d(mV)}{dt} = ma = \frac{W}{g}\frac{dV}{dt} \quad \text{(if W assumed constant)} \qquad (4)$$

and summing forces along the flight path:

$$\Sigma F = F - D - W \sin \gamma = \frac{W}{g}\frac{dV}{dt} \qquad (5)$$

or, rearranging:

$$\sin \gamma + \frac{1}{g}\frac{dV}{dt} = \frac{F - D}{W} \qquad (6)$$

noting that:

$$\sin\gamma = \frac{\left(\frac{dh}{dt}\right)}{V} \qquad (7)$$

substituting (7) into (6) and multiplying by V:

$$\frac{dh}{dt} + \frac{V}{g}\frac{dV}{dt} = \frac{V(F - D)}{W} \qquad (8)$$

Now notice that we have an expression for the airplane's excess power V(F - D). Comparing with (3), we define <u>specific excess power</u> as:

$$P_s \equiv \frac{V(F-D)}{W} = \frac{dE_h}{dt} = \frac{dh}{dt} + \frac{V}{g}\frac{dV}{dt} \qquad \left(\frac{ft - lb}{lb - sec}\right) \qquad (9)$$

<u>Level Acceleration Runs</u>

In acceleration tests, the first term of equation (9) can be eliminated by maintaining constant altitude ($\frac{dh}{dt} = 0$), and specific excess power can be measured in terms of airspeed and linear acceleration ($\frac{dV}{dt}$). The variation of specific excess power with airspeed is determined for one altitude over the entire level flight speed range of the airplane. This is done by conducting level acceleration runs from near minimum airspeed (V_{min}) to the maximum level flight airspeed ([illegible]) or maximum airspeed (V_{max}).

Practically, a constant altitude acceleration run cannot be flown. Imprecise pilot technique may cause oscillations about the desired altitude and changes in altimeter position error may result in a slight climb or descent during the acceleration; however, because potential and kinetic energies are both computed, the resultant total rate of change of specific energy is unaffected.

In other words, the airplane performance is considered in terms of total energy rather than potential or kinetic energy separately, and slight variations in altitude (potential energy) are considered perfectly convertible to airspeed (kinetic energy) or vice versa. The validity of this assumption depends upon smooth piloting technique and a small divergence from the assigned altitude. Abrupt longitudinal control inputs result in unaccountable energy losses, and large deviations from test altitude result in unaccountable changes in engine performance from that at the test altitude. The accuracy desired dictates the allowable excursions in normal acceleration and altitude during any run. In general, the allowable excursions increase with excess power available and decrease with increasing altitude (less excess power). Typical values to maintain P_s accuracy within approximately 5% are: $\pm$ 300 ft altitude excursion and $\pm$ 0.10g normal acceleration for an airplane with over 60 $\frac{\text{ft-lb}}{\text{lb-sec}}$ specific excess power.

Remember the two assumptions: no lift due to thrust and negligible gross weight change. The first of these will affect both level and climbing flight about equally. Since fuel consumption will be very small between data points recorded by the photopanel about every 2 sec, the assumption of negligible gross weight change should be valid for all current aircraft types (with the possible exception of supersonic, afterburning flight).

Sawtooth Climbs

In sawtooth climbs tests, the second term of equation (9) is eliminated by maintaining a constant true airspeed ($\frac{dV}{dt} = 0$), and excess power is measured in terms of vertical speed ($\frac{dh}{dt}$). The variation of specific excess power with airspeed can be determined for one altitude by conducting a series of short, constant true airspeed climbs at various airspeeds over the speed range of the airplane.

Since climb performance at given calibrated airspeeds is usually the data desired from sawtooth climbs, true airspeeds may not be held constant. For this case, true airspeed is increasing throughout the test climb which will result in actual P_s being greater than the indicated rate of climb.

A graph of the variation of rate of climb with airspeed for the test conditions is prepared from these data. A horizontal tangent to this curve defines the airspeed for maximum rate of climb at that altitude and gross weight. A tangent through the origin (V = 0, R/C = 0) determines the airspeed for maximum climb angle.

The observed climb performance is affected by changes in airplane gross weight and by variations in power or thrust available. A change in airplane weight has two effects on climb performance. First, the rate of climb and climb angle are inversely proportional to weight. In addition, a weight change causes variations in induced drag, which alters the power and/or thrust required curve, thus affecting the airspeeds at which maximum rate of climb and maximum climb angle occur. The change in these airspeeds is negligible for small variations around a standard (mission-related) gross weight. Variations in ambient temperature produce changes

in power (or thrust) available that are essentially constant with airspeed for propeller driven (jet) airplanes. Therefore, variations in power available will affect only the value of the rate of climb for propeller driven airplanes and not the airspeed at which it occurs. Similarly, the effect of variations in ambient temperature on the airspeed for maximum climb angle for turbojet airplanes is negligible.

When only the airspeeds for (and not the actual values of) maximum rate of climb and maximum climb angle are of interest, the effect of decreasing weight during the test may be minimized by obtaining the data in a particular order. If the constant airspeed climbs are flown in ascending order of airspeed, than in descending order at the same airspeeds, the average airplane gross weight for each pair of data points will be approximately equal. This technique may be further complemented by correcting each data point to an average test weight. A curve faired through all of the data will be representative of the climb performance at the average gross weight and the test day conditions (see Figure 7 for an example of such a graph).

Other factors which may affect the test results are the influence of vertical air currents and changing wind velocity with altitude (wind shear). Vertical air currents usually can be avoided by careful selection of test conditions. The effect of wind shear can be minimized by performing climbs on a heading which is perpendicular to the reported winds.

In general, high climb rates and excessive testing time render the sawtooth climb method impractical for turbojet airplanes except in the high lift and single engine configurations. The most significant advantage is that the data are collected in the climb attitude and the reduction in induced drag associated with climbs is accounted for.

TEST PROCEDURES AND TECHNIQUES-ACCELERATION RUNS

Preflight Procedures

a. Required data.

Time (or time increment).

V_o (continuous record during run).

H_{p_o} (continuous record during run).

OAT (start and end of run) or T_a.

Fuel used or fuel remaining (continuous record during run).

Photopanel is primary data source. Record correlating data on kneepad card.

b. Prepare da a card (Figure 2).

WEATHER			CARD NO. LP-1
ACCELERATION RUNS			PTR BIS TPS 75-70-1
AIRPLANE TYPE T-38A	BU. NO. 158197	TIME T.O. / TIME LAND	DATE 25 Nov 78
PILOT PIPE		T.O. C.G. GEAR DOWN % UP 26.2 %	
CONDITION NORMAL TRAINER - 1 pilot			T.O. GROSS WEIGHT 12,150 lb

EXTERNAL CONFIGURATION

NOSEBOOM, OAT probes

		start			END				
RUN	H_{po}	V_o	OAT_o	FUEL	V_o	OAT_o	FUEL	P/P	
1									
2									
3									
4									
5									
6									

FLIGHT DATA [illegible]

Figure 2
Sample Pilot's Data Card

Flight Procedures

a. Conduct a practice acceleration run at the test altitude commencing as near V_{min} as practicable. Use the acceleration run technique described in Section II and demonstrated during the performance demonstration flight.

b. Take data on next run.

c. Repeat procedure at other assigned altitudes.

DATA REDUCTION - ACCELERATION RUNS

General

Data reduction to standard conditions is normally performed by Computer Services Division (CSD). Data inputs are presented in Section XIII. This data reduction scheme accounts for the effects of nonstandard gross weight, ambient temperature (thrust), and induced drag changes.

For manual data reduction, follow the order presented below in setting up the data reduction form. Reduce the best acceleration run at each altitude assigned. Runs made near standard gross weight are preferred because the corrections will be smaller. Record elapsed time t, V_o, and H_{p_o} each 10 sec (at least 15 data points required). Record OAT_o for three data points.

Manual Determination of Specific Excess Power, P_s

Test day rate of change of specific energy corrected to a standard airplane weight will be determined for each data point. Rate of change of specific energy is analogous to specific excess power.

a. Correct V_o for instrument and position error to obtain V_c. Assume lag error negligible.

b. Correct H_{p_o} for instrument and position error to obtain H_p.

c. Determine Mach number corresponding to above values of V_c and H_p.

d. If necessary, calculate T_a (^oK) from OAT_o at three data points.

1. Correct OAT_o for instrument error.

2. Calculate M corresponding to time of observation of OAT.

3. Determine t_a by entering Appendix IV with OAT and M.

4. Calculate $T_a\ (^oK) = t_a\ (^oC) + 273$.

e. Calculate TAS for each data point. Use the average value of T_a determined in step d.

$$V_T = M\sqrt{\gamma g R T_a} = 65.7\ M\ \sqrt{T_a}\ (^\circ K) \qquad \text{(fps)}$$

f. Construct a working plot of the variation of V_T and H_p with elapsed time. Use scales of 100 fps/in. for airspeed, 200 ft/in. for altitude, and 20 sec/in. for time. Fair a curve through the data points. After fairing the data, you will observe that at an instant when the data shown is greater than an average rate of climb, the linear acceleration is less than average (see Figure 3 for an example of such a plot).

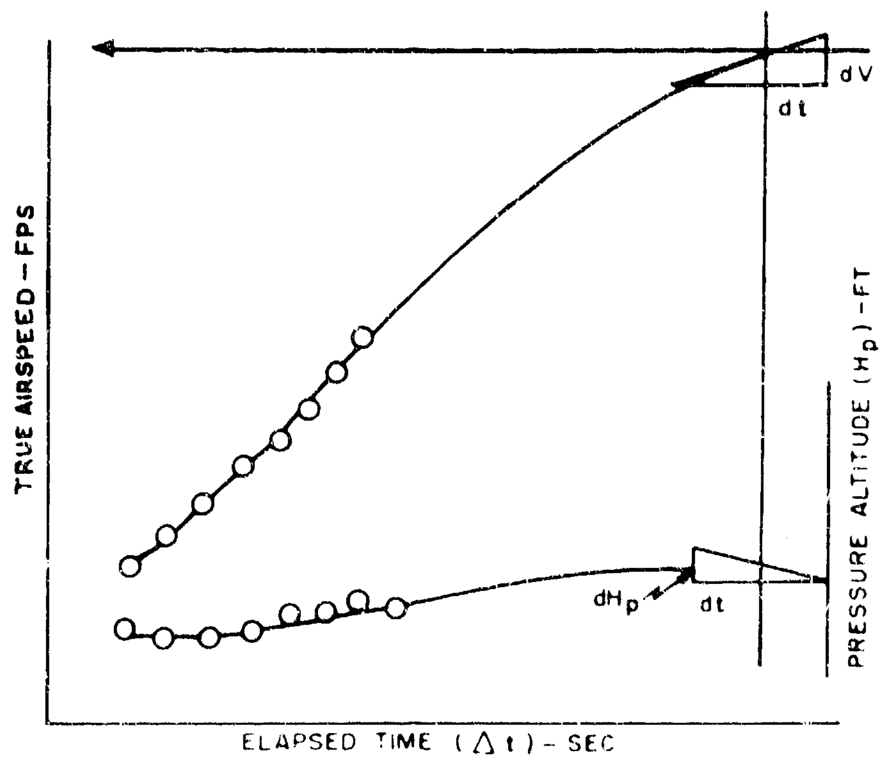


Figure 3
Acceleration Run Working Plot

g. Graphically measure dV/dt and dH_p/dt for at least eight points on the curves of step f. Record also the corresponding value of V_T. Caution - a protractor should not be used for measuring these derivatives.

h. Obtain standard day ambient temperature $(T_a)_s$ from ICAO atmosphere table. Calculate rate of change of geometric altitude, dh/dt. This correction is necessary because of nonstandard density.

$$\frac{dh}{dt} = \frac{dH_p}{dt} \times \frac{T_a}{(T_a)_s} \qquad \text{(fps)}$$

i. Determine airplane weight for the run. Use fuel data recorded at the end of the run.

j. Calculate ratio of airplane weight for test to standard airplane weight, W_t/W_s. Use the values of airplane standard weight listed in TPS Airplane Details.

Correction of P_s to Standard Conditions

In determining rate of change of specific energy for standard conditions, it is necessary to account for the differences in airplane weight from standard weight and ambient temperature at the test altitude from standard day ambient temperature. Differences in weight affect the momentum and the induced drag. These differences result in a different acceleration characteristic being observed for the test conditions than would have occurred for standard conditions. While the corrections for gross thrust and ram drag cannot be considered negligible (particularly near the service ceiling of the airplane), these corrections are tedious and will not be made manually. The induced drag correction is small in relation to the rest of the terms and will also be neglected for the manual data reduction. Both these corrections are included in the computer data reduction.

k. Calculate rate of change of specific energy for standard conditions. The gross weight correction is implicit in the equation.

[illegible]

i. Compute Mach number for each value of V_T obtained in step g.

$$M = \frac{V_T \quad (fps)}{65.7 \times T_a \ (^\circ K)}$$

Graphical Presentation of Results

a. Plot the variation of specific excess power for standard weight and ambient temperature with Mach number or calibrated airspeed (Figure 4).

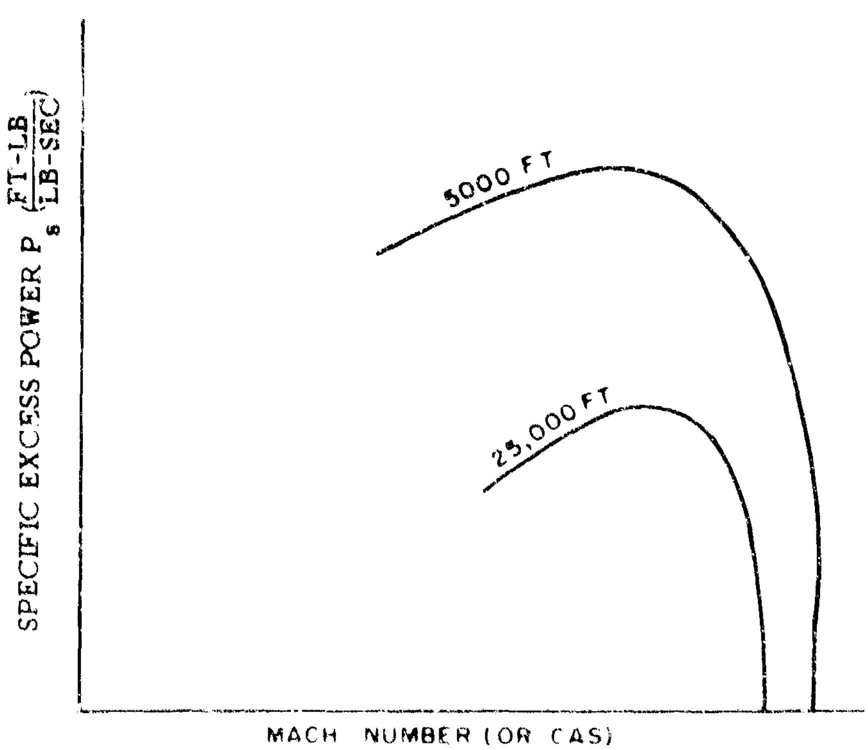


Figure 4
Graph of Excess Power Characteristics

b. When submitting the flight report, include the working plot of the variation of airspeed and altitude with elapsed time (step f).

TEST PROCEDURES AND TECHNIQUES -SAWTOOTH CLIMBS

Pertinent Particulars

a. Make several constant airspeed climbs (or descents) in ascending order of observed airspeeds from the minimum test airspeed to the maximum. Then make the same number of climbs in descending order at approximately the same airspeeds.

b. Use military or combat rated thrust in jets. Maintain constant power in propeller driven airplanes by adjusting MAP or torque as necessary during the climb. For the altitudes at which full throttle is reached attempting to maintain constant power, use the MAP or torque at the test altitude as an average value. Note the altitude at which full throttle is reached.

c. Photopanel data are desired; however, prepare the report on the basis of kneeboard data if photopanel data are not available.

Preflight Procedures

a. Required Data.

V_o.

Elapsed time for climb (Δt).

Altitude band (ΔH_p).

Fuel remaining.

OAT at test altitude (or T_a).

RPM and MAP or torque at test altitude.

b. Prepare data card (Figure 5).

WEATHER	CARD No. LP-1
SAWTOOTH CLIMBS	TPS-76-70-1
AIRPLANE TYPE: T-28B	BU. NO.: 151651
DATE: 15 MAR 78	
PIPE	GEAR DOWN / UP 25.5
CONDITION: NORMAL TRAINER (2 pilots)	T.O. GROSS WEIGHT: 8395

ALTITUDE BAND: 2000 - 4000 FT

RUN	V_o	Δt	W_f	OAT	RPM	MAP		
1	85							
2	110							
3	120							
4	130							
5	140							
6	160							
7	V_{max}							
8	160							
9	140							
10	130							
11	120							
12	110							
13	85							

SPEEDBRAKES

FLIGHT DATA CARD

Figure 5
Sample Pilot's Data Card

c. Review assigned scope of test and make notes on data card to be used as a reminder during the test, such as: configuration details, desired climb airspeeds, and altimeter barometric setting of 29.92 in Hg.

d. Obtain test altitude forecast winds from aerology.

e. Carry a 60-sec stopwatch.

f. Review photopanel operating procedures and switchology. For this test, use a camera speed of 1/2 frame per second.

g. Before leaving the flight line, check the operation of the photopanel.

Flight Procedures

The sawtooth climbs method consists of a series of constant observed airspeed climbs through an altitude band. All climbs are performed on the same heading perpendicular to the forecast wind at the test altitude.

a. At 300 to 500 ft below the altitude band on the required climb heading, trim the airplane in level flight at the desired climb airspeed.

b. Set the power or thrust and establish a stabilized climb. Check configuration details and trim the airplane. Maintain constant power by adjusting MAP or torque frequently in the climb.

c. Energize the photopanel about 10 sec before entering the altitude band.

d. With the stopwatch, measure the time required to transit through the altitude band or 2 min, whichever is less.

e. When passing the test altitude, mentally note the OAT, MAP (or torque), and RPM readings.

f. About 10 sec after passing out of the altitude band, de-energize the photopanel, level off, and record data (Figure 6).

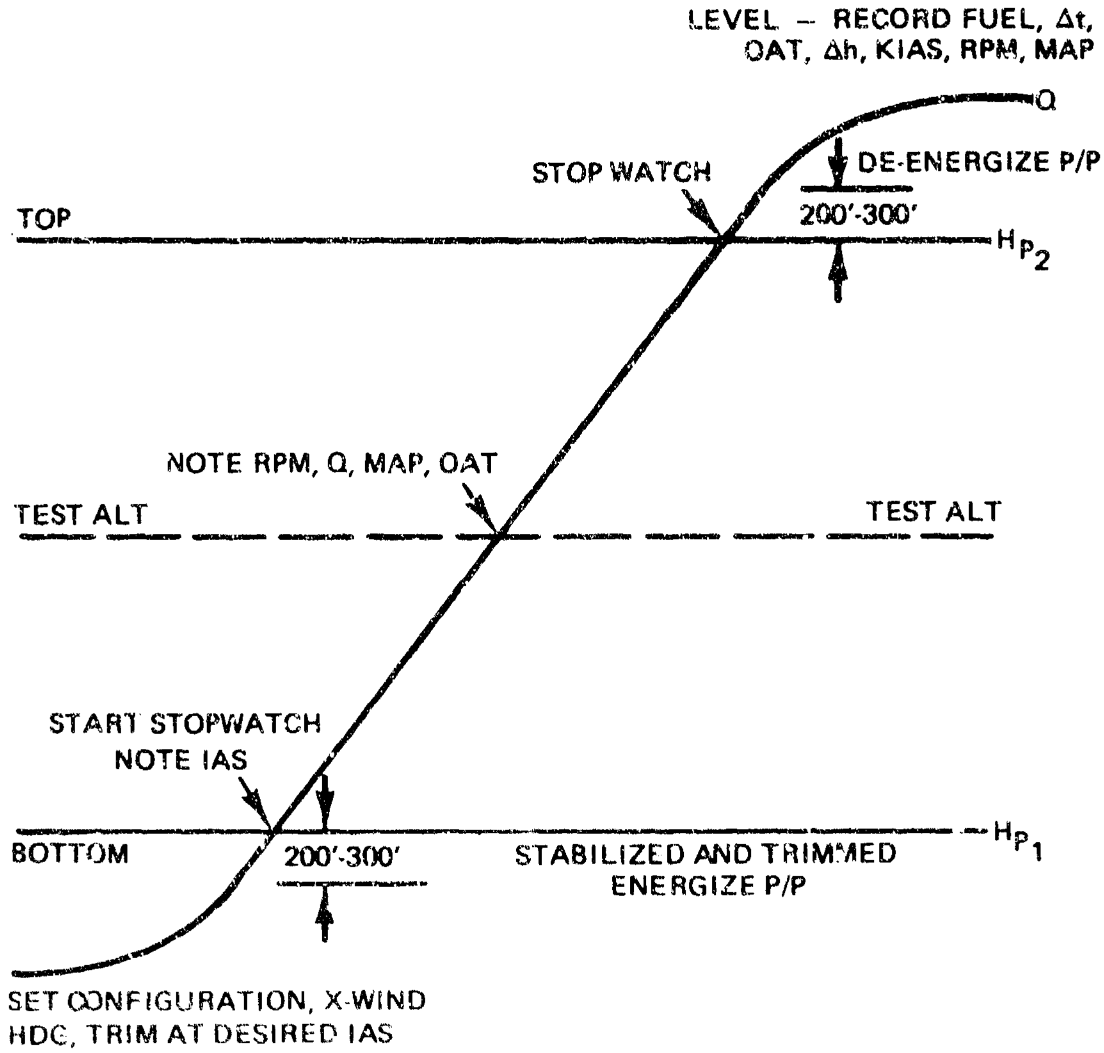


Figure 6
Sawtooth Climb Flight Procedures

g. Reverse course, descend, and repeat the procedure at the next desired airspeed.

h. Since the rate of climb will be zero, the maximum airspeed run must be made at the test altitude. Ensure that the airplane has stabilized in airspeed before recording data.

i. It is not imperative that the climb airspeeds be precisely the planned airspeed for the climb. For good results, the airspeed at the upper limit of the band must be identical with the airspeed at the lower limit. In the climb, attempt to maintain airspeed within ± 1 kt of the initial value.

j. If the wind at the test altitude is not strong, it is preferred to make the climbs in a racetrack pattern. In stronger wind conditions, use an "S" pattern with all turns upwind.

DATA REDUCTION - SAWTOOTH CLIMBS

General

Sawtooth climb data will be reduced on the school's Hewlett-Packard computers, which make corrections to standard airplane gross weight and to standard atmosphere conditions (thrust correction). Review the appropriate program folder to determine data requirements and input format.

Transcribe the following data on a data reduction sheet.

V_o, $H_{p_{o_1}}$, $H_{p_{o_2}}$, Δt, RPM or torque, MAP, w_f, and OAT at mid-altitude.

Manual Determination of Airspeeds for Highest Rate of Climb and Maximum Climb Angle

a. Correct observed airspeed and altitude for instrument error. Assume negligible difference between the altimeter position error at the lower altitude and the error at the upper altitude.

$$V_i = V_o + V_{ic}$$

$$H_{p_i} = H_{p_o} + H_{pic}$$

b. Calculate.

$$\Delta H_{p_i} = H_{p_{i_1}} - H_{p_{i_2}}$$

c. Calculate indicated rate of climb.

$$R/C_i = \frac{\Delta H_{pi}}{\Delta t} \quad \text{fpm}$$

d. Calculate airplane gross weight for each data point.

e. Calculate average test gross weight

$$W_A = \frac{W \text{ for each test point (sum)}}{\text{number of runs}}$$

f. Calculate weight correction factor for each data point

$$\frac{W_t}{W_A}$$

g. Ignoring the induced drag correction, corrected rate of climb for each data point may be calculated

$$R/C_c = \left(\frac{W_t}{W_A}\right) \quad (R/C_i)$$

This approximation is not always valid for rates of descent.

h. Correct at least three observed OAT values for instrument error.

i. Determine Mach number at which each OAT value in step h was obtained. Enter Appendix II with V_i and mid-altitude.

j. Determine ambient air temperature from Appendix IV.

k. Construct a graph of variation of corrected rate of climb with indicated airspeed. Fair one curve through the data. This curve shows the variation of corrected rate of climb with airspeed for the average gross weight of the test and test conditions (see Figure 7).

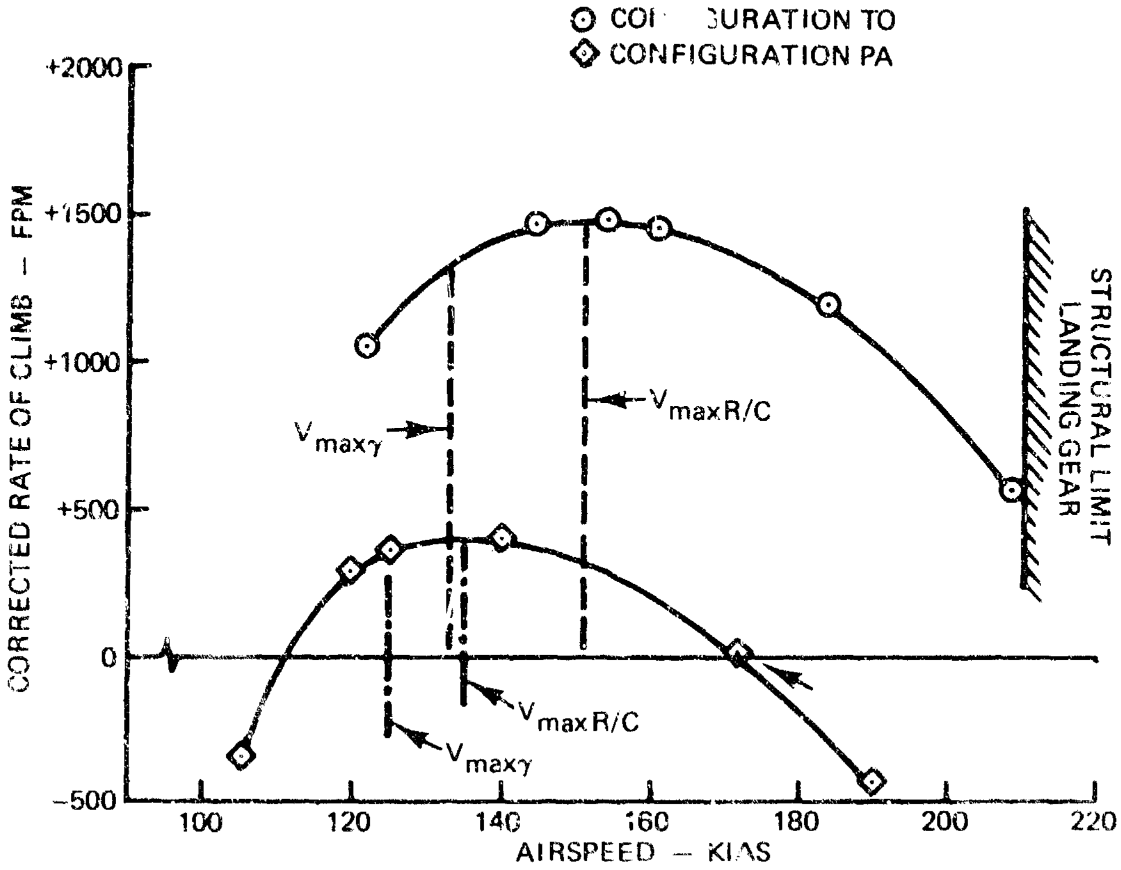


Figure 7
Sample of Graphical Presentation of Results

SECTION VII

LEVEL FLIGHT PERFORMANCE

REFERENCES

Section VII

1. AGARD, VOL I, Chapter 4; pp 4:1-4:23; Chapter 6, Sec 6.2.
2. Petersen, *Aircraft and Engine Performance*, Chapters 4, 5, and 6.

NOTATIONS INTRODUCED IN SECTION VII-A

F_g/δ	Engine referred gross thrust (a function of the engine exhaust nozzle pressure ratio)
w_f	Fuel flow, pph
F_n/δ	Referred net thrust, lb
$w_f/\delta\sqrt{\theta}$	Referred fuel flow, pph
$N/\sqrt{\theta}$	Referred engine speed, percent of rated RPM
W/δ	Referred gross weight, lb
W_F	Fuel weight

NOTATIONS INTRODUCED IN SECTION VII-C

SHP_c	Chart shaft horsepower determined from the manufacturer's engine operating chart for the conditions of the test.
SHP_t	Test shaft horsepower actually delivered by the engine for the test conditions. Determined from correction of SHP_c or from engine torquemeter installation.
$\Delta SHP_{R/C}$	SHP correction for rate of climb or descent.
BHP_{rt}	Brake horsepower required for stabilized, level flight (i.e., airplane drag) for the test conditions (gross weight, pressure altitude, and ambient air temperature).
SHP_{ew}	Equivalent shaft horsepower required referred to a standard airplane gross weight.
V_{md}	Airspeed for minimum drag.
$\frac{w_f}{\delta\sqrt{\theta}}$	Referred fuel flow.
$\frac{SHP}{\delta\sqrt{\theta}}$	Referred shaft horsepower.
$RNI \; \frac{\delta}{\theta\sqrt{\theta}}$	Reynold's No. Index

SECTION VII-A

JET AIRPLANE LEVEL FLIGHT PERFORMANCE

PURPOSE OF TEST

The purpose of this test is to evaluate the level flight performance of a jet airplane by:

a. Determining the thrust required and/or fuel flow required.

b. Determining the referred gross thrust or fuel flow available.

c. These data will be determined for one value of referred gross weight (W/δ) and will subsequently be correlated with similar data for other W/δ values to determine the level flight performance over the operating range of the airplane.

DISCUSSION AND THEORY

The thrust required for level flight of an airplane may be described in terms of the functional expression

$$\frac{F}{\delta} = f\left(M, \frac{W}{\delta}\right) \qquad (1)$$

This expression was derived from a theoretical analysis of level flight performance (drag equation). It suggests a flight test procedure for experimentally determining the thrust required characteristics; maintain a constant value of W/δ in flight and measure the referred thrust required at several values of Mach number. If we repeat this procedure for several values of W/δ and construct a suitable graph of these results, the thrust required for the airplane may be determined for any combination of airplane gross weight, pressure altitude, and Mach number. A flight test procedure of this form is particularly suitable for jet airplanes because the

changes of airplane drag caused by Mach number effects are accounted for, and a wide range of airplane gross weight can be analyzed in only a few flights. But thrust is difficult to measure and correct to standard conditions (EPR gauge would be useful), and we are more concerned with range and endurance data. In order to evaluate range and endurance, we must define the combined performance of the airframe and the engine. By dimensional analysis of the factors affecting jet engine performance, we can show that:

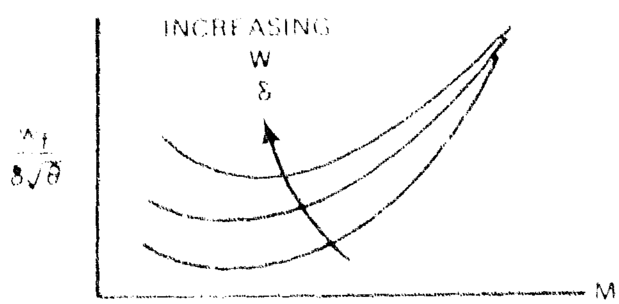
$$\frac{w_f}{\delta\sqrt{\theta}} = f(M, \frac{N}{\sqrt{\theta}}) \tag{2}$$

$$\frac{F}{\delta} = \frac{D}{\delta} = f(M, \frac{N}{\sqrt{\theta}}) \tag{3}$$

These relationships describe the installed jet engine. Combined with the airframe relationship (Equation (1)) the generalized equation for the total airplane in level flight is

$$\frac{w_f}{\delta\sqrt{\theta}} = f(M, \frac{W}{\delta}) \tag{4}$$

Now we can fly our airplane at constant ratios of $(\frac{W}{\delta})$, measure (referred) fuel flow when stabilized in level flight at several Mach numbers, and define curves describing the performance of the airplane over a wide range of gross weights and altitudes.



Two assumptions were made: no lift due to thrust (high α) and negligible Reynold's number effects.

In level flight, referred gross weight decreases as fuel is used. A constant value of W/δ may be maintained by increasing pressure altitude at a predetermined schedule as airplane gross weight decreases. In obtaining a series of level flight thrust required data points each at the same value of W/δ, the altitude must be increased for each successive data point with the airplane arriving at each altitude with sufficient excess fuel to stabilize in airspeed before the desired W/δ occurs. The flight path may be visualized as a series of ascending steps.

TEST PROCEDURES AND TECHNIQUES

Pertinent Particulars

a. Use constant altitude technique for data points at airspeeds above that for minimum drag and constant airspeed (thrust adjustment) techniques for airspeeds near and below minimum drag where stabilization at constant thrust is difficult.

b. Obtain sufficient data to define airframe and engine characteristics for one value of W/δ. Concentrate on defining the minimum thrust (fuel flow) and maximum range portions of the curve, if preliminary data are available.

c. It is recommended that the V_{max} data point at assigned W/δ be taken first and the remainder of the W/δ data be obtained in order of decreasing airspeeds.

d. Determine the thrust or fuel flow available at the assigned reference pressure altitude using an acceleration run technique. Obtain data from 0.7 V_{mrt} to the maximum level flight airspeed. For the lowest W/δ values, the thrust available data should be obtained first in order to reduce the airplane's gross weight and thus increase the initial W/δ test altitude.

Preflight Procedures

a. Obtain W/δ assignment.

b. Prepare pilot's data card. In flight, at each stabilized airspeed at the assigned W/δ value, record the following data on the data card and the photopanel.

Airspeed (V)

Pressure altitude (H_p)

Fuel flow (w_f)

Engine RPM (N)

Engine Pressure Ratio

OAT or T_a

Fuel remaining (W_F) or fuel used

Photopanel counter number

c. Construct a W/δ card. The W/δ card is a kneeboard size graph which shows a variation of pressure altitude (corrected for instrument error) with fuel. It is used to maintain a constant referred gross weight ($\frac{W}{\delta}$). The card is constructed as follows: for at least four fuel weight values between the appropriate initial airplane weight at altitude and 1,000 lb fuel remaining:

1. Calculate airplane gross weight, W.

2. Calculate δ required to maintain assigned W/δ

$$\delta = \frac{W}{W/\delta}$$

3. Obtain corresponding pressure altitude from interpolation of ICAO atmosphere tables.

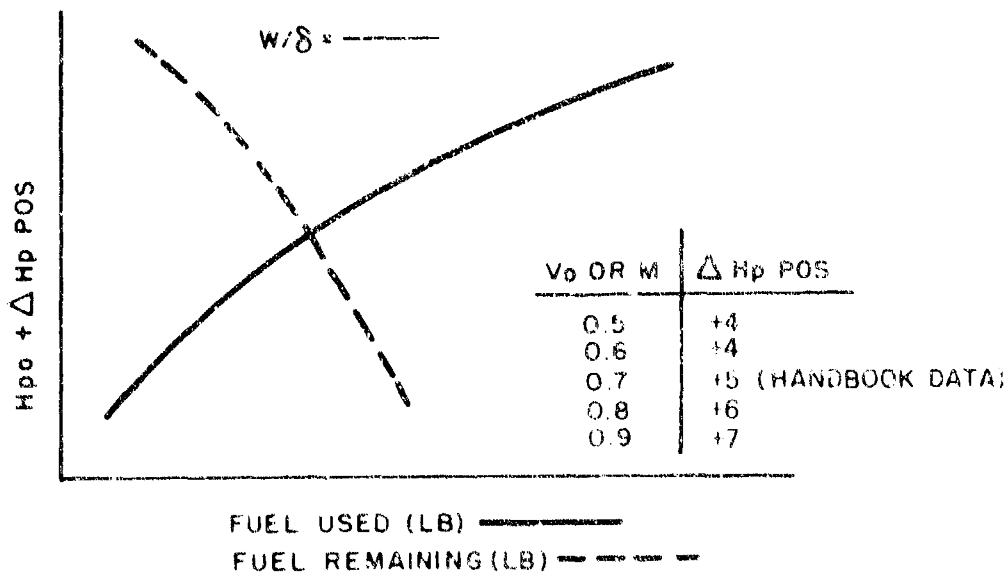
4. Compute

$$H_{p_o} + \Delta H_{p_{pos}} = H_{p_c} - \Delta H_{p_{ic}}$$

5. For several values of Mach number at the average pressure altitude of the test, determine the altimeter position error correction.

6. Construct a graph of the variation of $H_{p_o} + \Delta H_{p_{pos}}$ with fuel used. Include altimeter position error information in a table on the graph.

d. Use of W/δ card. Estimate the fuel used or remaining value for the next data point, enter the graph, and obtain the required pressure altitude. Subtract from this figure the approximate position error and stabilize the airplane at this computed observed pressure altitude. An example of such a graph is:



e. A simple and easy method of determining the observed pressure altitude required to maintain an assigned W/δ value is to have radio contact with a flight test engineer who can compute the required altitude as the airplane gross weight decreases during the flight. The pilot informs him of the estimated fuel and the airspeed at the next data point. The engineer computes the airplane gross weight and the pressure altitude required to maintain the assigned W/δ at that weight. He then subtracts the altimeter position error and instrument error and informs the pilot of the required observed pressure altitude.

Flight Procedures

a. Constant Altitude Technique (Stable Equilibrium Conditions)

1. This technique requires the pilot to estimate the fuel required to stabilize at each data point. The amount of fuel required will be dictated by: (a) engine/airplane characteristics, (b) weather conditions, and (c) pilot proficiency. For constant altitude points, 5 min or 300 lb fuel can be used as an initial allowance for stabilization until the pilot becomes more familiar with the requirements. At the first point, subtract the estimated fuel required from the present fuel reading; the resulting figure will be the fuel weight reading at the data point. From the W/δ card determine the observed pressure altitude required to maintain the assigned W/δ at that fuel reading. Record the computed observed pressure altitude and fuel reading for reference.

2. Climb to the computed observed pressure altitude, set the thrust, and allow the airspeed to stabilize. Military Rated Thrust is recommended for the first data point (V_{mrt} point).

3. Check configuration.

4. Precisely maintain the computed altitude until the reference fuel reading is obtained. If the airspeed is stabilized, record the data. If not stabilized, record data anyway, but set up a new point at the same thrust.

5. Estimate fuel weight at the next data point and repeat steps 2, 3, and 4.

b. Constant Airspeed or Thrust Adjustment Techniques (Unstable Equilibrium Conditions)

1. At airspeeds less than the airspeed for minimum drag, where stabilization at a constant altitude and thrust setting is difficult, use the constant airspeed technique.

2. Determine the reference fuel reading and pressure altitude as in the constant altitude technique.

3. At the required altitude, maintain the desired airspeed and adjust the thrust to maintain level flight. The task is made more difficult by the necessity of attaining the required test altitude. A certain degree of pilot skill and patience is required to do this well. First set a thrust which results in a climb at the trim airspeed. Then reduce the thrust to obtain a descent. At the required altitude, adjust thrust to obtain level flight or a slight climb. Continue these "bracketing" corrections until stabilized at the required altitude and airspeed.

4. When the reference fuel reading is obtained, record data.

c. If using a fuel used system, record fuel remaining at the last W/δ data point also. Later, calculate gal fuel remaining and add to final fuel used reading to check accuracy of fuel used counter.

d. Determination of thrust available. Before or after obtaining the required W/δ data, fly to the assigned reference altitude. Stabilize at approximately 0.7 V_{mrt} with MIL thrust set and the speedbrakes partially extended. When the engine thrust has stabilized, energize the photopanel, leave MIL thrust set, and retract the speedbrakes. As the airplane accelerates in level flight, engine pressure differential, airspeed, pressure altitude, and OAT or T_a will be recorded out to V_{mrt}. These data define a curve of the variation of MIL thrust with airspeed for the test day conditions. For engines without thrust measurements, referred fuel flow data will be used. These data will later be corrected to standard day conditions to determine standard day V_{mrt} (correlated with thrust required data).

e. The recommended airspeed for endurance is sometimes determined from minimum acceptable flying qualities rather than minimum fuel flow. During the flight, record the airspeed which you would consider the minimum desirable from the standpoint of satisfactory handling qualities.

DATA REDUCTION

General

Airframe (drag) and engine (fuel flow) characteristics will be determined in terms of the variation of referred gross thrust, referred engine speed, and referred fuel flow with Mach number for one value of W/δ. Because this presentation of results is generalized, data obtained on different days but in the same airplane should be compatible. Normally, data reduction is performed on the Hewlett-Packard computer with the first and last data points worked manually. Record the following data on the data reduction sheet:

V_o, H_{p_o}, Engine Pressure Ratio, w_{f_o}, W_F, N_o, OAT or T_a

Data Reduction - Thrust required (Omit steps d, e, and f if thrust data not available

a. Correct observed airspeed and pressure altitude for instrument and position error.

b. Determine Mach number from $V_c - M - H_p$ chart.

c. Determine δ. Interpolate from ICAO Standard Atmosphere table.

d. Calculate ambient air pressure.

$$P_a = 29.92\ (\delta) \qquad \text{(in. Hg)}$$

e. Calculate and/or correct engine exhaust total pressure (P_{T5}) or (P_{T7}) for instrument error and calculate P_T/P_a.

f. Enter calibration curve for test engine with P_T/P_a to obtain corresponding F_g/δ.

g. Correct observed fuel flow for instrument error.

h. Calculate total fuel flow in lb/hr. Assume density of JP-5 fuel is 6.8 lb/gal and JP-4 fuel is 6.5 lb/gal.

i. Correct observed OAT for instrument error and determine ambient air temperature from Appendix IV.

j. Calculate

$$\sqrt{\theta} = \sqrt{\frac{t_a + 273}{288}}$$

k. Calculate airplane gross weight.

l. Calculate:

referred fuel flow $\left(\frac{w_f}{\delta\sqrt{\theta}}\right)_{Test}$

referred gross weight $\left(\frac{W}{\delta}\right)_{Test}$

m. Construct the following graphs for the average value of W/δ flown.

Airframe (drag) Characteristics (F_g/δ vs. M)

(omit for airplanes without thrust measurement)

Airplane Characteristics ($w_f/\delta\sqrt{\theta}$ vs. M)

n. On the graph of the variation of referred gross thrust required with Mach number also plot the variation of referred gross thrust (available from the acceleration run) at the reference altitude. For airplanes in which inflight thrust was not measured, plot the variation of referred fuel flow available at the reference altitude on the graph of the variation of referred fuel flow required with Mach number. The intersection of these curves defines the maximum airspeed at the reference altitude and at the airplane's gross weight for the test day ambient

temperature. Correction of the thrust available (or fuel flow available) data to standard temperature as discussed in Section VII-B will give the standard day maximum level flight airspeed.

o. From the faired curve of $w_f/\delta\sqrt{\theta}$ vs. M, read values of referred fuel flow for several values of Mach.

p. Determine the ambient air pressure ratio, δ, at the reference altitude from the standard atmosphere tables, Appendix I.

q. Calculate specific range for each value of Mach (step o).

$$S.R. = \frac{661M}{\delta\left(\frac{w_f}{\delta\sqrt{\theta}}\right)} \qquad \text{nmi/lb}$$

r. Construct a graph of S.R. vs. M for the value of W/δ flown.

s. Using the equation

$$w_f = \delta\sqrt{\theta}\,(w_f/\delta\sqrt{\theta}) \qquad \text{pph}$$

the standard day values of actual fuel flow may be plotted against V_o.

SECTION VII-B

DETERMINATION OF STANDARD DAY MAXIMUM LEVEL FLIGHT AIRSPEED

General

The maximum level flight airspeed of a thrust limited jet airplane is determined by the intersection of its thrust available and thrust required curves for one altitude. In Section VII-A, methods for determining standard day thrust required and test day thrust or fuel flow available were determined. The thrust or fuel flow of a jet engine at MIL thrust varies with altitude, airspeed, and ambient temperature. Standard day thrust or fuel flow for the test altitude can be computed by making an incremental correction for the temperature difference from standard based on estimated or previously determined engine characteristics. If the engine is not operating at rated RPM, a similar thrust or fuel flow correction can be made for the nonstandard RPM setting. Corrections to engine thrust and fuel flow data are usually imprecise. However, large corrections to inflight engine parameter measurements should be treated with caution. It is important to realize that more rigorous and tedious data reduction procedures exist.

Determination of Standard Day Thrust Available

a. For the several values of MIL thrust available discussed in Section VII-A, calculate standard day referred gross thrust available.

$$\left(\frac{F_g}{\delta}\right)_s = \left(\frac{F_g}{\delta}\right)_t + \left(\frac{d\ F_g/\delta}{dt}\right) (T_a - T_{a_s})$$

Obtain values of $\frac{d(F_g/\delta)}{dT}$ from graphs similar to Figure 1 for the appropriate engine.

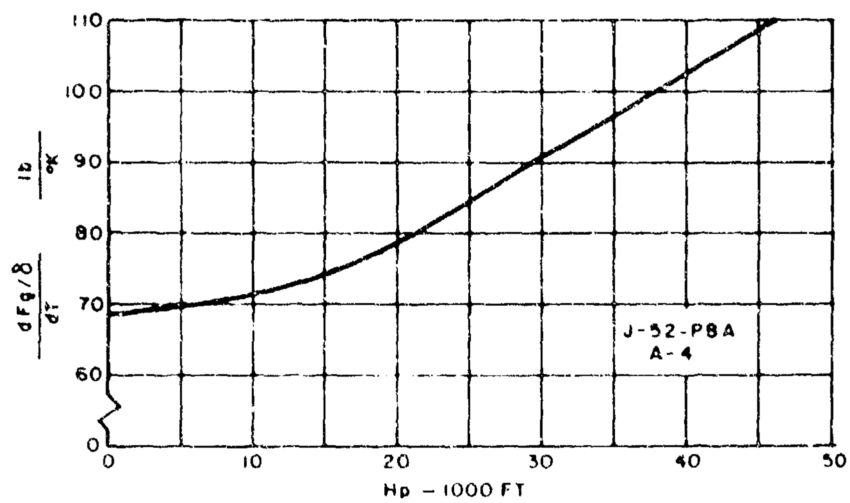


Figure 1
Engine Performance

b. Plot the standard day referred gross thrust available for each altitude on the graph of referred gross thrust required. Read the standard day maximum Mach number and calculate the corresponding true airspeed.

Determination of Standard Day Fuel Flow Available

Tailpipe pressure ratio data from which to compute F_g/δ sometimes is not available. Referred fuel flow, based on total temperature and pressure conditions $(w_f/\delta_T\sqrt{\theta_T})$, is a function of throttle setting, OAT, and Mach number. Since the throttle setting is constant (MIL), and since Mach effects are generally small, it is possible to plot $w_f/\delta_T\sqrt{\theta_T}$ vs. OAT for one throttle setting. With this graph of referred fuel flow available, the test day data may be corrected to standard day conditions. The following procedure is used:

a. For MIL thrust data points discussed in Section VII-A calculate

$$\frac{w_f}{\delta_T \sqrt{\theta_T}} = \left(\frac{w_f}{\delta \sqrt{\theta}} \right)_{Test} \times \frac{1}{(1 + .2M^2)^4}$$

for each engine separately.

b. For each engine plot $\frac{w_f}{\delta_T \sqrt{\theta_T}}$ for the MIL thrust points vs. OAT. Fair one curve to average data from both engines.

c. For the reference altitude at the standard temperature, determine the standard day OAT at several Mach numbers between .7 V_{mrt} and 1.1 V_{mrt} for the test day.

d. With the curve generated in step b, determine referred fuel flow available for each OAT calculated in step c.

e. Determine standard day referred fuel flow available based on ambient temperature and pressure from

$$\left(\frac{w_f}{\delta \sqrt{\theta}} \right)_s = \left(\frac{w_f}{\delta_T \sqrt{\theta_T}} \right) (1 + .2M^2)^4 \quad \text{(double for 2 engines)}$$

f. Plot the referred fuel flow available (step e) vs. Mach number on the referred fuel flow required curve in Section VII-A. The intersection of the fuel flow required and available curves is the standard day V_{mrt}.

SECTION VII-C

POWER REQUIRED AND FUEL CONSUMPTION

PURPOSE

The purpose of this test is to determine the generalized power required characteristics for a propeller driven airplane and to determine the variation of referred fuel flow with referred shaft horsepower of a turboprop engine for one altitude. These characteristics are required to describe the level flight performance of the airplane.

DISCUSSION AND THEORY

General

In describing airplane level flight performance, the parameter of greatest interest is specific range, the ratio of true airspeed to fuel flow. Crudely, specific range may be determined by flying the airplane at a constant airspeed and altitude and measuring the corresponding fuel flow. The level flight performance for a large range of airspeeds, altitudes, and gross weights is more efficiently evaluated by investigating separately the airframe characteristics (power required for level flight) and the engine characteristics (fuel flow variation with power and altitude).

Dimensional analysis of the level flight performance of a low speed airplane shows that for a given loading and configuration, the variation of equivalent shaft horsepower required with equivalent airspeed is a single curve which is applicable to all altitudes of airplane operation. This generalization is valid if a constant parasite drag coefficient, propeller efficiency, and effective aspect ratio can be assumed. Although this generalization is valid for the cruise airspeed range of

most propeller driven airplanes, the assumptions fail at very high angles of attack (where effective aspect ratio and propeller efficiency decrease markedly) and at high airspeeds where transonic effects may be observed. This generalization is a convenient method of fairing test data because for a given airplane gross weight we may write

$$(BHP_{ew}V_{ew}) = K_1 + K_2 \; (V_{ew}{}^4)$$

where the subscript, w, signifies values for a standard gross weight. This is the equation of a straight line when plotted in terms of the variables enclosed in parentheses. Figure 1 is an example of such a plot.

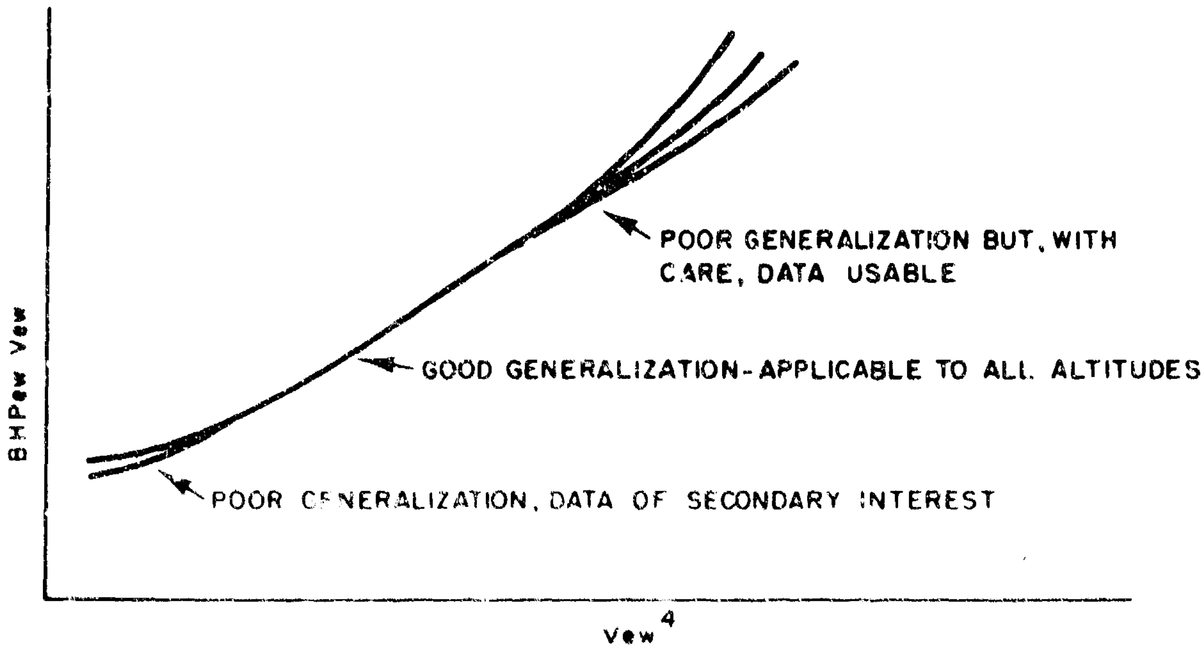


Figure 1
Generalized Power Required

In the power required test, it is necessary that the power and fuel flow data be representative of stabilized flight conditions. The data must be obtained in smooth air which is free of vertical air currents. At airspeeds greater than the airspeed

for minimum power required, stabilization in airspeed and altitude is easily obtained. At lower airspeeds, data may be obtained by accepting a small vertical velocity (less than 50 fpm) at constant airspeed and later calculating a correction for the incorrect power setting. The power required for level flight at a given airspeed varies with airplane gross weight and flight data must be corrected to a standard gross weight.

<u>Turboprop Engine Data Generalization</u>

The relationship between fuel flow and shaft horsepower output for a turboprop engine can be expressed by the following functional relationship:

$$\frac{w_f}{\delta\sqrt{\theta}} = f\left(\frac{SHP}{\delta\sqrt{\theta}},\ M,\ RNI\right)$$

Flight tests with current turboprop engines have shown that the Mach (and RNI) effects on the above relationship are weak or not significant for the normal operating envelope of the engine, and thus the data can be generalized using the following simplified relationship:

$$\frac{w_f}{\delta\sqrt{\theta}} = f\left(\frac{SHP}{\delta\sqrt{\theta}}\right).$$

If this form fails to generalize the data, good generalization may be obtained by using total vice static conditions, i.e.,

$$\delta_T = \delta\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}}$$

$$\text{and}\quad \sqrt{\theta_T} = \sqrt{\theta}\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{1}{2}}$$

TEST PROCEDURES AND TECHNIQUES

Pertinent Particulars

a. For reciprocating engines, set the power needed to obtain the desired airspeed but keep BMEP high by maintaining a minimum RPM and maximum MAP within the handbook limits.

b. Use the constant altitude technique at airspeeds faster than the airspeed for minimum drag (V_{md}) and the constant airspeed technique at airspeeds slower than V_{md}.

c. It is recommended that the V_{mrt} data point be taken first and the remainder of the data be obtained in order of decreasing airspeeds.

d. The photopanel will not be used for this flight.

Preflight Procedures

a. Review engine limits, single engine operation, and airstart procedures.

b. Prepare data cards. In flight at each data point, record:

Run number

Pressure altitude

Airspeed

OAT/CAT

Torque/MAP

Fuel Flow

Fuel remaining

RPM/EGT

Configuration

Flight Procedures

a. Perform a series of stabilized, level flight runs at the assigned altitude using various power settings. Although a particular sequence of data is not required, it is recommended that the V_{mrt} point be obtained first and the remaining data be obtained in order of decreasing airspeeds.

b. For the constant altitude method, set the maximum cruise power setting and allow the airplane to stabilize in airspeed while maintaining a constant altitude. Several minutes at constant power and altitude are necessary to obtain good stabilization of airspeed. When certain of stabilization, record data. Reduce power setting to that estimated to give the next desired airspeed and wait for airspeed to stabilize.

c. For the constant airspeed method, establish a trimmed, constant airspeed and adjust the power until level flight is achieved at the test altitude. A bracketing technique, similar to the one described in Section II and demonstrated on the performance demonstration flight, should be used. If level flight cannot be achieved after several power adjustments, acceptable data can be obtained with the airplane in a climb or descent of 50 fpm or less. When this condition is established, record data and commence a 5 min run at constant airspeed and power setting. After 5 min record data again to obtain the average rate of climb and average power for this interval. The data will later be corrected for this condition.

d. The airspeed at which the airplane exhibits minimum acceptable flying qualities for maximum flying should be recorded for later use in determining the recommended endurance airspeed.

DATA REDUCTION

General

Follow the order of the data reduction presented below in setting up the data reduction form. Normal data reduction is by Hewlett-Packard Computer program but the first and last data points must be done manually.

Manual Data Reduction - Fuel Flow and Generalized Power Required for Turboprop Airplane

a. Correct observed pressure altitude for instrument and position error.

$$H_{p_c} = H_{p_o} + \Delta H_{p_{ic}} + \Delta H_{p_{pos}}$$

b. Correct observed airspeed for instrument and position error.

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$

c. Calculate equivalent airspeed. Use Appendix II for compressibility correction.

$$V_e = V_c + \Delta V_c$$

d. Correct observed OAT for instrument error.

e. Determine M by use of M - V_c - H_p chart.

f. Determine ambient air temperature (T_a) using Appendix IV (Assume K = 0.95).

g. Determine $1/\sqrt{\sigma}$ by entering Galcit chart with H_p and T_a.

h. Determine δ from PTM standard atmosphere tables and H_{p_c}.

i. Compute $\sqrt{\theta}$ from $\sqrt{\theta} = \left(\frac{T_a + 273°}{288}\right)^{1/2}$

j. Compute $\delta\sqrt{\theta}$

k. Correct observed torque and RPM for instrument error for one engine.

l. Compute test shaft horsepower

$$SHP_t = (K)\,(Q)\,(RPM)$$

where K is the engine torquemeter constant, Q is torque or pressure (psi or ft/lb) and RPM is propeller RPM.

m. Repeat steps k and l for the other engine.

n. Compute total test shaft horsepower (sum of both engines).

o. Compute airplane test gross weight (W_t).

p. For constant airspeed data, compute rate of climb. Assume that the altimeter instrument and position error corrections remain constant over the test interval.

$$R/C \doteq \frac{\Delta H_{p_o}}{\Delta t} \quad \text{fpm} \quad (\underline{+}50 \text{ fpm maximum})$$

or

$$R/C = \frac{\Delta H_{p_o}}{\Delta t}\left(\frac{T_{a\ test}}{T_{a\ std}}\right) \text{ fpm} \quad (T_{a\ std} \text{ at test } H_p)$$

q. Calculate shaft horsepower correction for rate of climb.

$$\Delta SHP_{R/C} = \frac{R/C\,(Wt)}{33{,}000\ \eta_p}$$

Assume $\eta_p = 0.85$ (an average value)

r. Compute shaft horsepower required for stabilized, level flight for the test conditions.

$$SHP_{rt} = SHP_t + \Delta SHP_{R/C}$$

s. Compute equivalent shaft horsepower required.

$$SHP_e = \frac{SHP_{rt}}{1/\sqrt{\sigma}}$$

t. Compute ratio of standard gross weight to test gross weight W_s/W_t and the correction factors $(W_s/W_t)^{1/2}$ and $(W_s/W_t)^{3/2}$.

u. Correct equivalent shaft horsepower required and equivalent airspeed to the standard gross weight.

$$SHP_{ew} = SHP_e\left(\frac{W_s}{W_t}\right)^{3/2}$$

$$V_{ew} = V_e\left(\frac{W_s}{W_t}\right)^{1/2}$$

v. Compute V_{ew}^4 and $V_{ew} \times SHP_{ew}$.

w. For each engine compute referred shaft horsepower =

$$\frac{SHP_t}{\delta\sqrt{\theta}}$$

x. For each engine, correct observed fuel flow for instrument error and compute referred fuel flow.

$$\frac{W_f}{\delta\sqrt{\theta}}$$

Manual Data Reduction - Fuel Flow and Generalized Power Required - Reciprocating Engine Airplane

a. Perform steps a through g listed under turboprop data reduction.

b. Correct observed RPM and MAP for instrument error.

c. Determine chart shaft horsepower from engine chart. The chart method is not as accurate as the engine torquemeter which should be used if available.

d. Obtain standard day ambient air temperature for test altitude from standard atmosphere tables.

$$\left(T_{a_s}\right)_k = T_{a_s} + 273 \qquad °K$$

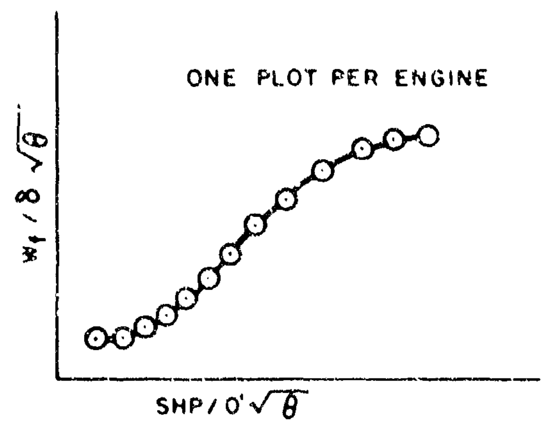
e. Correct observed CAT for instrument error.

f. Compute test day shaft horsepower

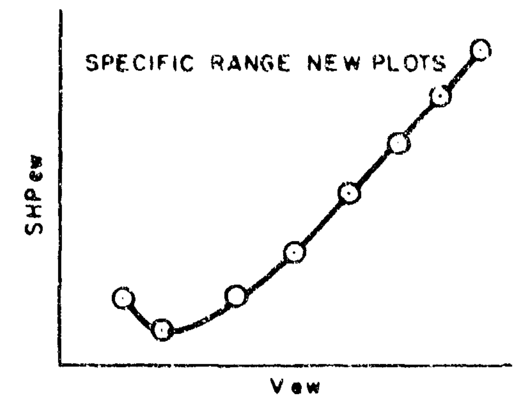
$$SHP_t = SHP_c \sqrt{\frac{T_{a_s}}{CAT_t}} \qquad °K$$

g. Perform steps o through v.

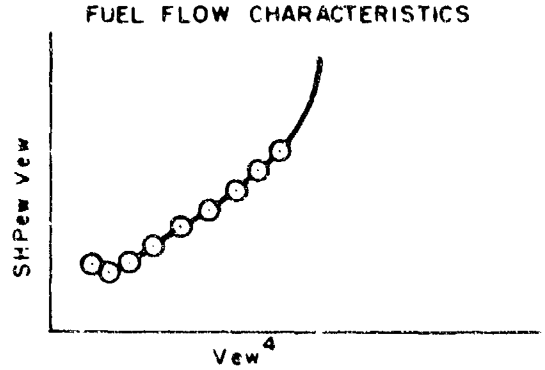
Graphical Presentation of Results



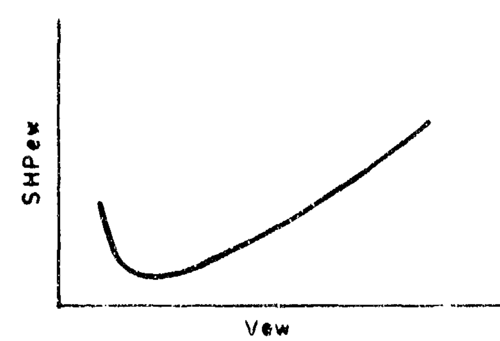
FUEL FLOW CHARACTERISTICS



GENERALIZED POWER REQUIRED



GENERALIZED POWER REQUIRED



COMPUTED POWER REQUIRED

SECTION VIII

MANEUVERING PERFORMANCE

REFERENCES

Section VIII

1. USAF ARPS PTM: Chapter IV of Techniques

   Chapter V of Theory

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MANEUVERING PERFORMANCE

INTRODUCTION

Aircraft maneuverability can best be defined as the ability to change direction and/or magnitude of the velocity vector. While this definition is accurately described and clearly understood by most pilots, the best method to optimize aircraft maneuverability is more difficult to define due to the complexity and dynamic nature of the combat maneuvering situation. Pilot selection of turning maneuvers, in terms of the change of direction achieved as a function of energy state, is strongly dependent on many factors, not the least of which is the energy state and position of the opponent aircraft or his weapons system. Assuming parameters such as pilot proficiency, roll performance, flying qualities, field of view, and fuel management techniques are equal, we have learned from experience that the offensive advantage is available to the airplane which initially has and can maintain the highest energy level. In addition, the airplane with the highest energy level has a better opportunity to engage and disengage at will. To achieve or deny this advantage depends primarily on two factors:

a. The weapons system capability.

b. The aircraft maneuvering performance.

Quantitatively, the weapons system capability can be depicted by release or firing envelopes, and once these initial conditions are known, the optimization problem becomes one of maneuvering into the effective weapons envelope. Such maneuvering is dependent upon the ability of the pilot to control turn, altitude, airspeed, and acceleration characteristics.

PURPOSE

The purpose of this section is to discuss aircraft maneuverability in terms of sustained and instantaneous g available, turn rate, turn radius, and the interface of these performance parameters with specific excess power characteristics.

DISCUSSION AND THEORY

A measurement of the relative tactical value of high performance airplanes is the maximum normal acceleration available at a given airspeed and altitude. It is important to distinguish between sustained g available which can only be achieved while maintaining a constant energy level, and instantaneous g available which can be achieved while gaining or losing energy at a given rate.

Sustained g Available

Sustained g available is determined by level flight turning performance. When an airplane is in straight, level flight, it develops lift equal to its weight. When this airplane is banked into a level turn, the lift increases so that the vertical component of lift remains equal to the airplane weight. Because the lift has increased, the drag has increased also, and the engine thrust must be increased to maintain the airspeed. The maximum bank angle for which both airspeed and altitude can be maintained is a function of both the aerodynamic and thrust characteristics but it is usually limited by the thrust available. If this maximum bank angle (or normal acceleration) is measured at several airspeeds, the sustained turning performance is determined.

Instantaneous g Available

Instantaneous g available is determined by the variation with Mach number of the normal acceleration for onset, tracking, and limit buffet. As an airplane is maneuvered more and more vigorously, a normal acceleration will be attained at which buffeting of the airframe is first detected (onset buffet). As the normal acceleration is increased further, the buffeting increases in intensity until it becomes so severe as to prevent satisfactory accomplishment of an offensive maneuver (tracking buffet). When the normal acceleration is increased still more, a maximum buffet acceptable for defensive operation of the airplane is observed (limit buffet). The variation with Mach number of the normal acceleration at which a specified level of buffet is observed is known as a buffet boundary, and for the specified condition, this buffet boundary defines the operational V-n envelope of the airplane (Figure 1).

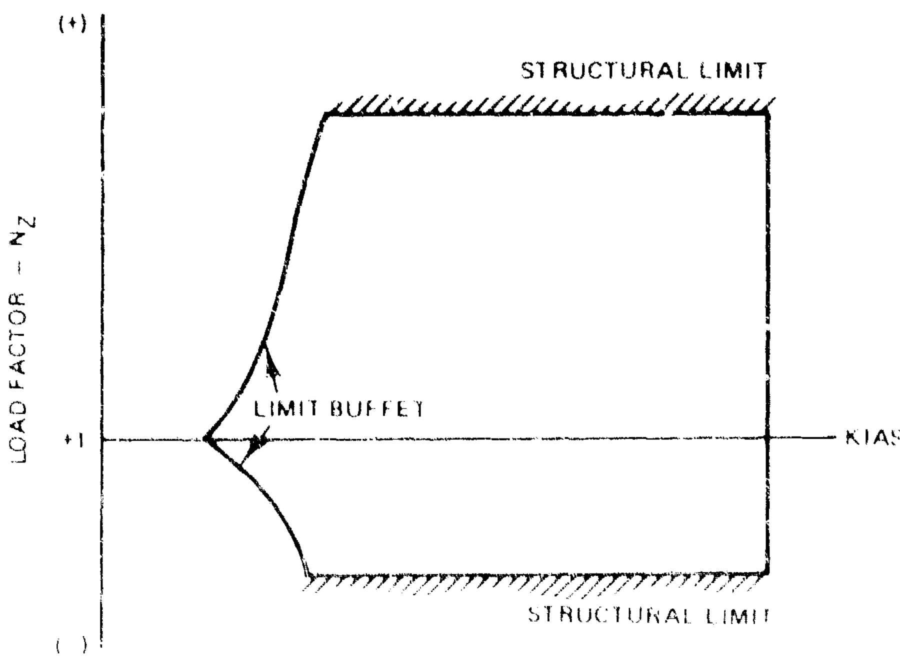

Figure 1
Typical V-n Diagram

For most flight conditions, the limit buffet level may be defined by: the structural limit, aerodynamic limit ($C_{L_{max}}$), longitudinal control limit, or maximum load factor for acceptable flying qualities of the airplane.

In all cases, buffet levels must be defined in relation to the mission of the airplane. Some airplanes (transports, straight-wing trainers, etc.) may have no offensive or defensive maneuvers related to their mission which bear upon instantaneous turning performance. These airplanes may exhibit little or no buffeting prior to reaching their aerodynamic limit (stall); or they may be prohibited from buffeting flight due to comfort or structural safety reasons. Onset and limit buffet may be coincident; and there may be no relatable or applicable tracking buffet level for an airplane.

Airframe buffet is associated with the disturbance of airflow somewhere on the airplane's structure. The intensity of the disturbance will vary with angle of attack and Mach number. For a given Mach number, a specified buffet level will occur at the same angle of attack (or lift coefficient). Figure 2 depicts buffet characteristics typical of a supersonic fighter airplane.

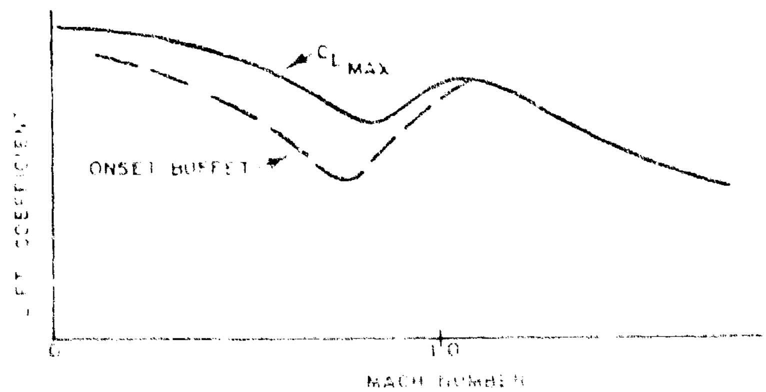

Figure 2
Buffet Characteristics of a Fighter Airplane

Rate and Radius of Tur

For a given normal acceleration, n_z, at a given altitude, the radius of turn is given by

$$R = \frac{V_T^2 \text{ (fps)}}{g\sqrt{N_z^2 - 1}} \qquad \text{ft}$$

and the rate of turn is given by

$$\omega = \frac{57.3\, V_T}{R} = \frac{g\sqrt{N_z^2 - 1}}{V_T} \qquad \text{deg/sec}$$

Turn rate is perhaps the most tactically significant of all the parameters used to measure turn performance. It measures rate of change of flight path direction as a function of sustained or instantaneous g available and airspeed. In the air-to-air maneuvering environment, the relative turn rate capability of two adversaries will directly influence the tactics used by each combatant. Of course, turn radius (which also is a function of TAS and g available) must be considered when determining the mission suitability of an airplane. An example of the use of turn radius is the gun tracking solution in which a certain amount of lead (smaller turn radius) is required.

Tactical Considerations

Another important consideration in maneuvering performance is the use of the vertical gravity vector to effectively add to the airplane's lift vector. The effect of increasing bank angle on turn rate is shown in Figure 3. The requirements of the mission will dictate acceptable load factors, turn rates, and radii. Consideration must be given to required maneuvers, visual contact distances, performance of potential adversaries, pilot load factor tolerance, and the expected altitude at

which various maneuvers will be conducted. The influence of flying qualities, energy level maintenance, and weapons systems must also be considered when evaluating the turning performance of an airplane.

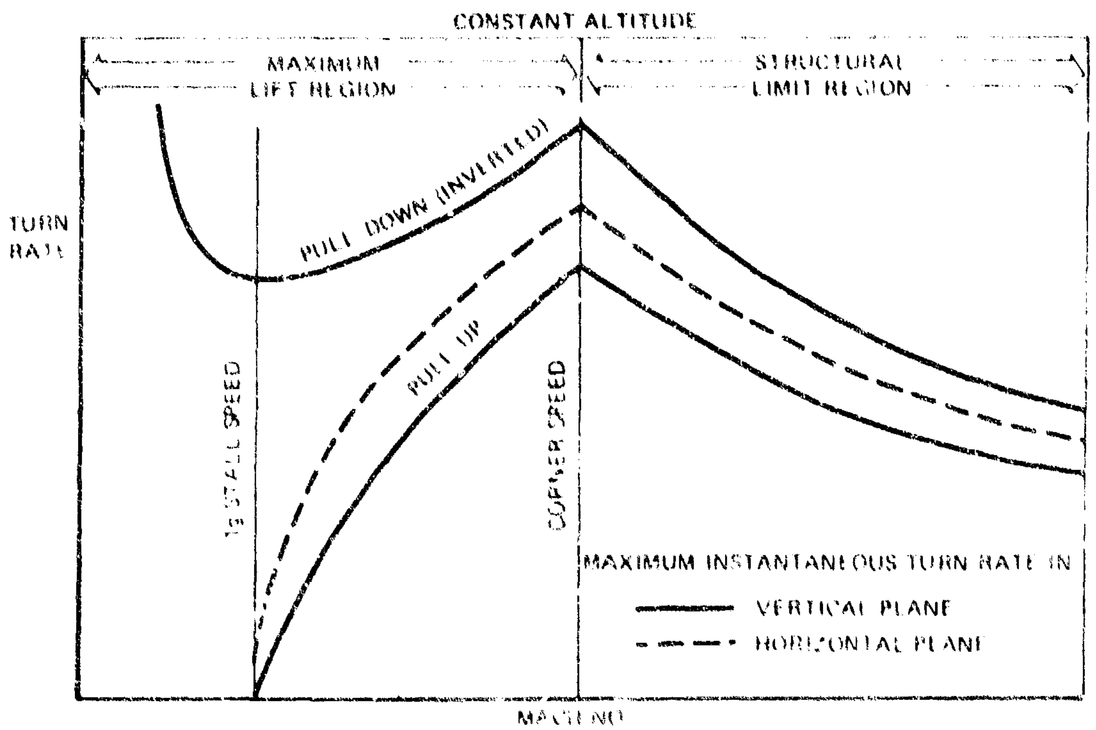

Figure 3
Three-Dimensional Turn Comparison

Wing Loading and Thrust-to-Weight Considerations

Assuming there is no appreciable lift due to thrust, the relationship of wing loading (W/S) on turn performance may be expressed

$$L = C_L \, 1/2 \, \rho V_T^2 \, S$$

where

$$L = W \cdot n_z \text{ (available instantaneous)}$$

Rearrangement of this expression results in

$$n_z \underset{\text{inst}}{\text{avail}} = \frac{C_L \, \rho V_T^2}{2 \, (W/S)}$$

At any given altitude and airspeed, instantaneous g available is solely dependent upon wing loading (W/S) and the airplane lift coefficient (C_L).

The effect of thrust-to-weight ratio (T/W) on turn performance may be expressed in stabilized, unaccelerated flight as

$$T = D$$

$$T = (D/L) \, L$$

where $L = W \cdot n_z$ (available instantaneous)

Rearrangement of this expression results in

$$n_z \underset{\text{sust}}{\text{avail}} = (T/W) \, (C_L/C_D)$$

The sustained n_z available is directly proportional to the thrust-to-weight ratio (T/W).

Sustained g available is also a function of aerodynamic (airframe) characteristics through its relationship with the lift-to-drag ratio (C_L/C_D).

A graphical presentation of the variation in sustained and instantaneous turn rate with wing loading and thrust-to-weight ratio is shown in Figures 4 and 5, respectively. At a constant altitude and Mach, the sustained turning performance in Figure 4 is increased with higher thrust-to-weight ratios and improved aerodynamic characteristics (reduced wing loading). However, at the same constant altitude and Mach, the instantaneous turning performance in Figure 5 is increased with lower wing loading and is unaffected by the thrust-to-weight ratio.

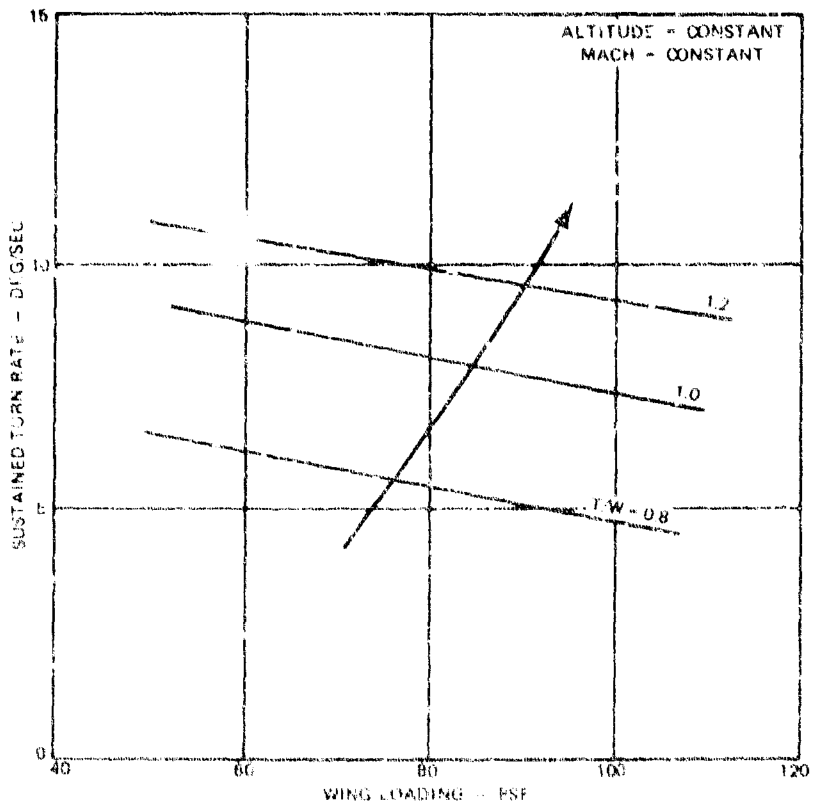

Figure 4
Effect of (T/W) on Sustained Turning Performance

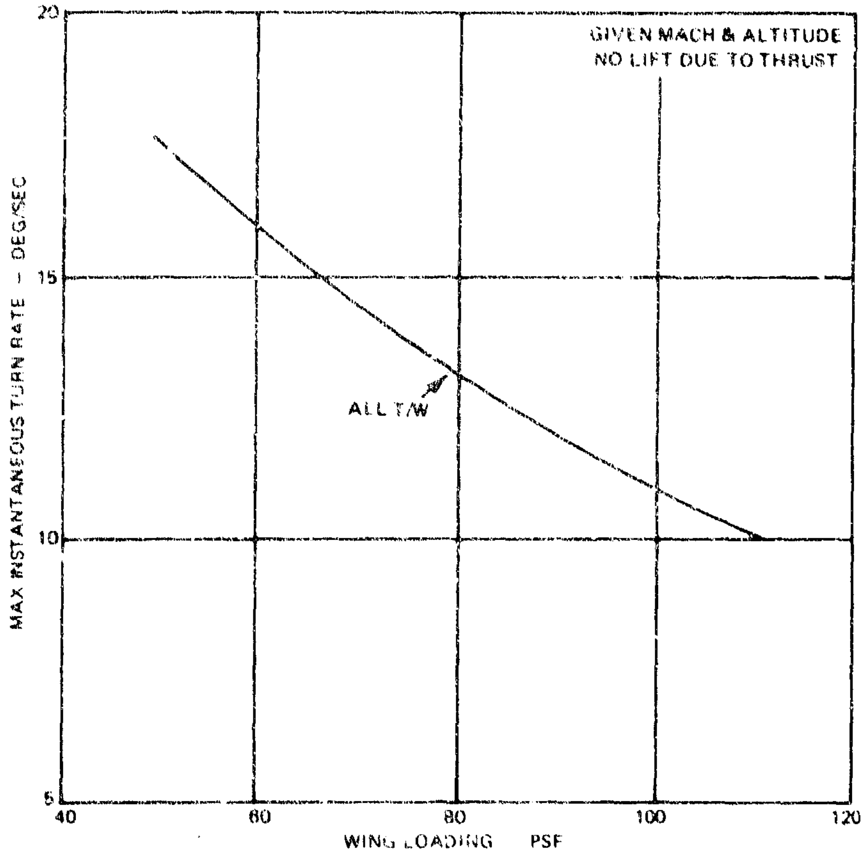

Figure 5
Effect of (W/S) on Instantaneous Turning Performance

Energy Maneuverability

Thus far, aircraft maneuverability has been explained in terms of directional change (turn) only, but magnitude changes of the velocity vector are equally important. From the specific energy equation

$$E_h = h + \frac{V^2}{2g}$$

it can be deduced that SUSTAINED maneuverability is closely related to specific energy levels in terms of altitude and airspeed. By observing the correlation of energy rate variation with turning performance, tactical maneuverability can be related to the amount of energy possessed and how well that energy is managed.

A comparison of specific excess power characteristics (P_s) can be used to determine the operational flight envelope in which an airplane can gain or lose energy more quickly than an adversary. The amount of energy gained or lost is directly related to maintaining maneuvering position against an adversary aircraft. The envelope for optimum energy maneuverability may be represented on an altitude vs. Mach number (H-V) diagram containing contour lines of constant specific excess power (Figure 6). Energy gain per unit time is maximum at the points where the lines of constant energy altitude (E_h) are tangent to the lines of constant P_s. Flight along the maximum energy path will allow the most efficient energy conversion of altitude and airspeed.

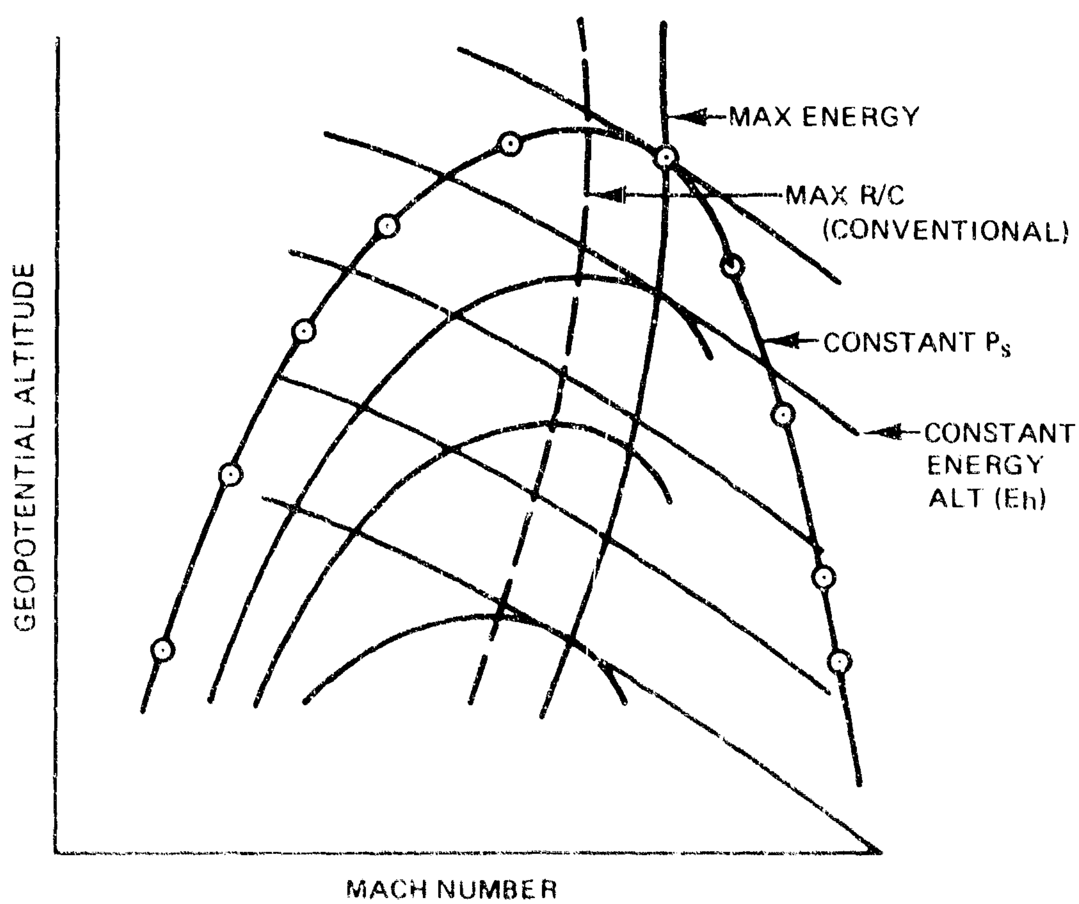

Figure 6
H-V Diagram

The actual technique to determine areas of numerical specific excess power advantage involves the overlay of constant P_s lines of one aircraft over another. This technique is illustrated in Figure 7. The shaded area represents the flight region where aircraft A has maneuvering advantage over aircraft B. One step further in the analysis of energy maneuverability would be to determine contours of constant differential specific excess power (ΔP_s) as shown in Figure 8, the shaded area represents the flight region where aircraft A has its greatest potential for maneuvering advantage. These energy rate diagrams can also be used to determine areas of sustained maneuvering advantage as a function of g.

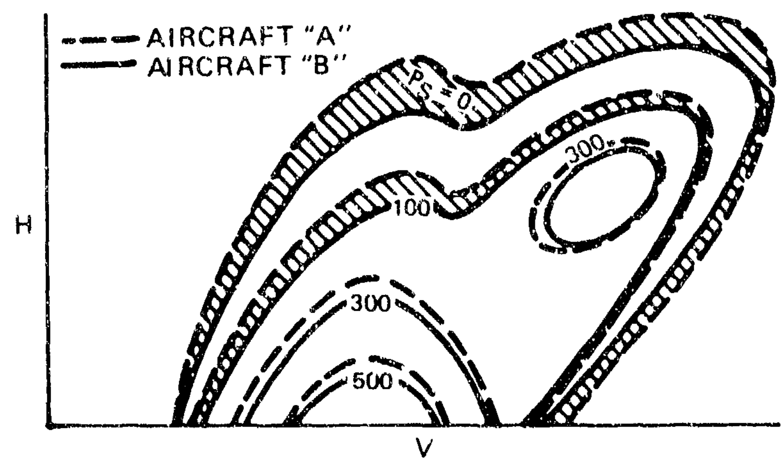

Figure 7
Specific Excess Power Overlays

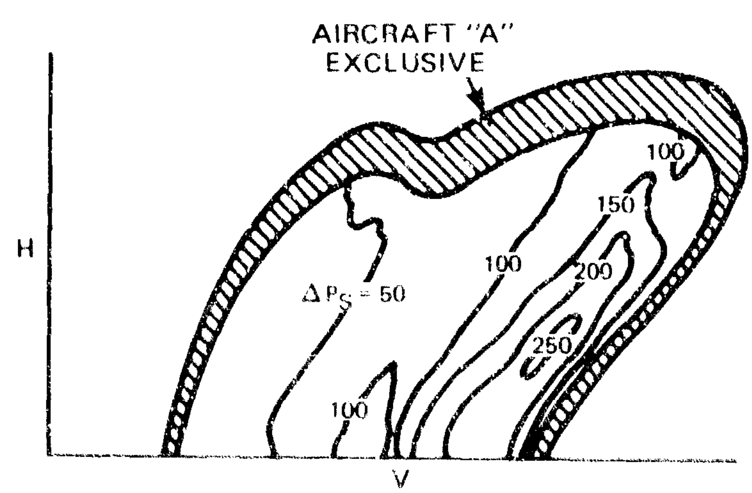

Figure 8
Differential P_s Contours

Since energy rate diagrams (P_s) are not limited to 1g, a maximum maneuver corridor may be represented by an area between the maximum energy path at 1g and the maximum energy path at the aerodynamic or structural g limit. By employing the overlay technique at various g levels, areas of tactical advantage for all flight conditions can be determined between adversary aircraft (Figure 9).

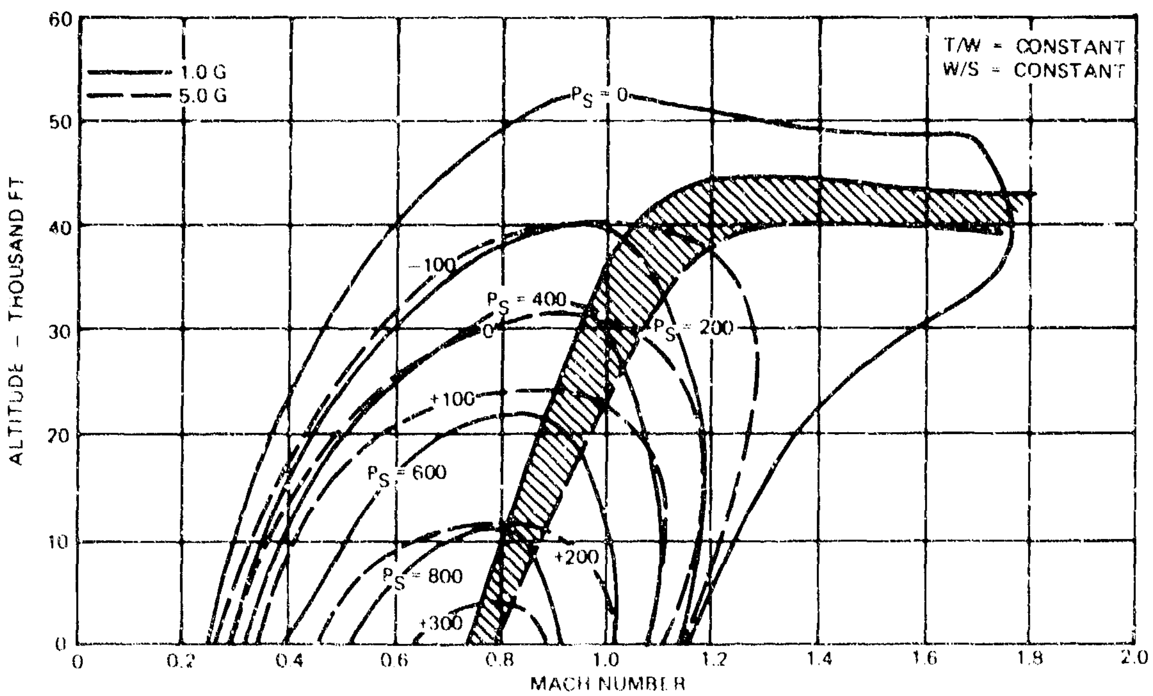

Figure 9
Maneuver Corridor

The energy rate variation from a turn rate of zero (1g) to the maximum turn rate achievable (aerodynamic or structural g limit) characterizes the complete spectrum of energy maneuverability for any given airspeed and altitude combination. The typical variation illustrated in Figure 10 provides three useful measures of maneuvering performance. The 1g energy rate indicates a measure of acceleration and/or climb performance. The turn rate at zero energy rate ($P_s = 0$) is that which can be sustained without energy loss. The maximum instantaneous turn rate corresponds to the highest energy loss rate. By improving the lift characteristics and/or reducing the drag characteristics, the energy rate variation with turn rate will be less proncunced and a general fanning out of the curve will result (Figure 11).

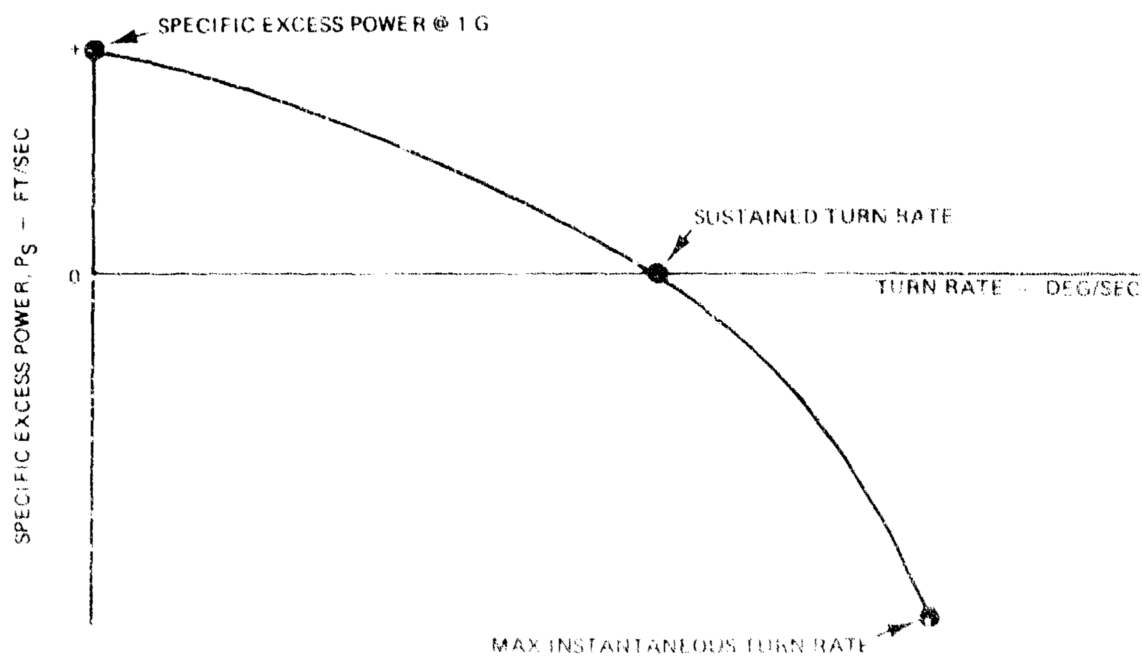

Figure 10
Energy - Maneuverability

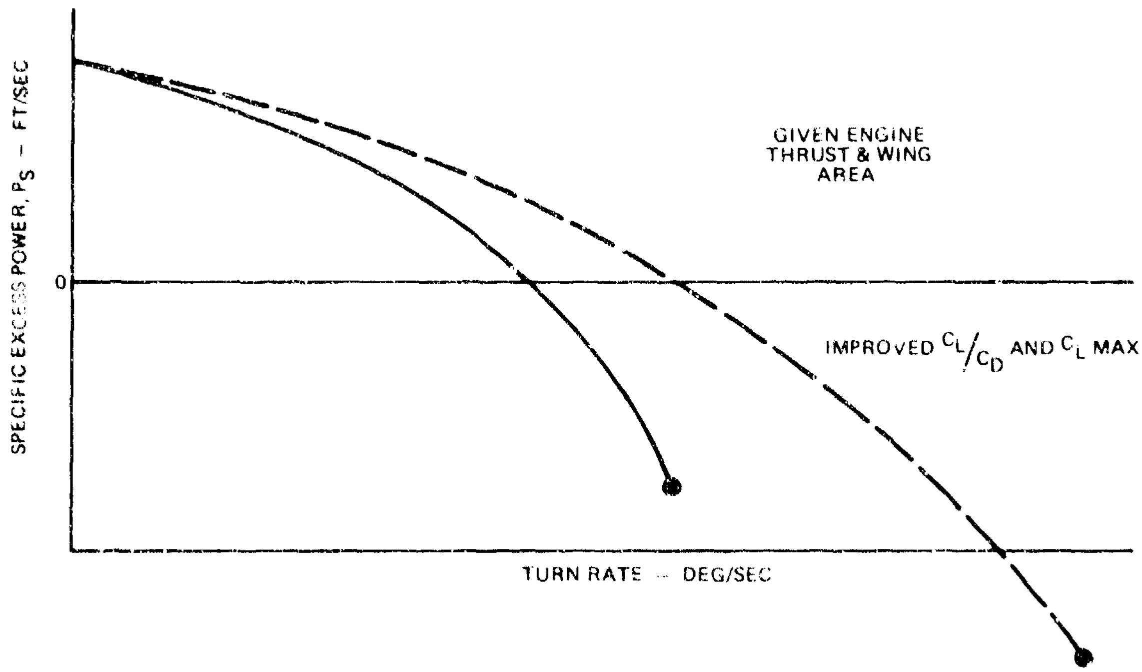

Figure 11
Influence of Improved Aerodynamics

The effect of wing loading on the energy maneuverability curve is illustrated in Figure 12. The greatest benefit of reduced wing loading occurs at and near to the maximum instantaneous performance limit. The influence of thrust-to-weight on the energy maneuverability curve is illustrated in Figure 13. Increasing the thrust-to-weight ratio has no significant effect on the instantaneous turn performance, but significantly improves the 1g and sustained maneuvering capabilities.

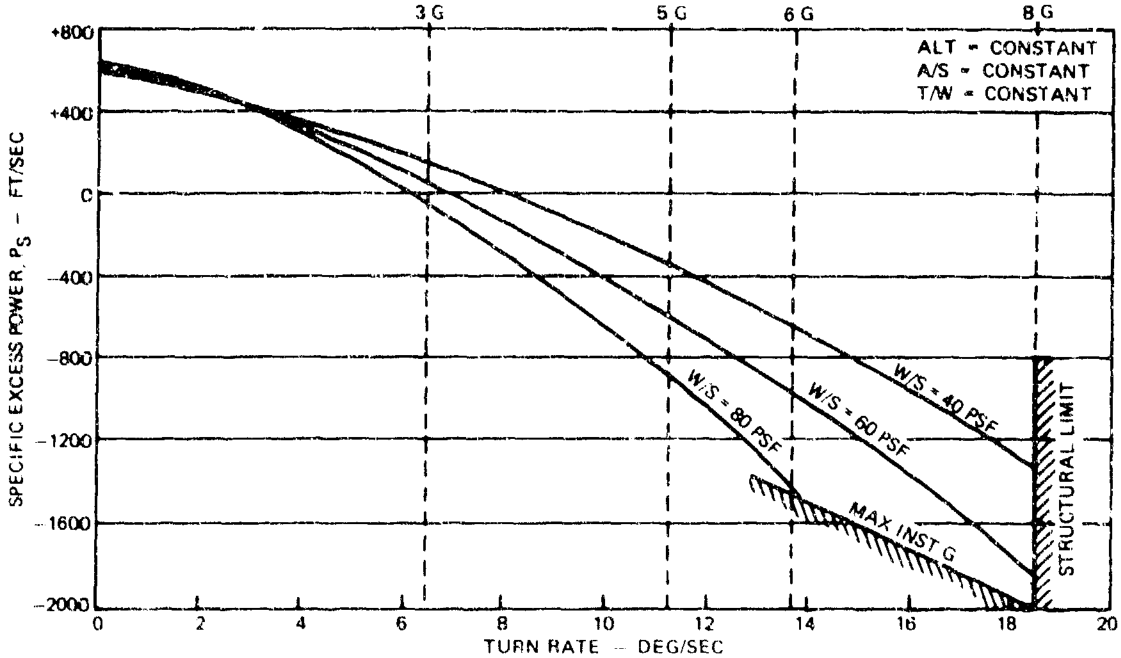

Figure 12
Effect of Reduced Wing Loading

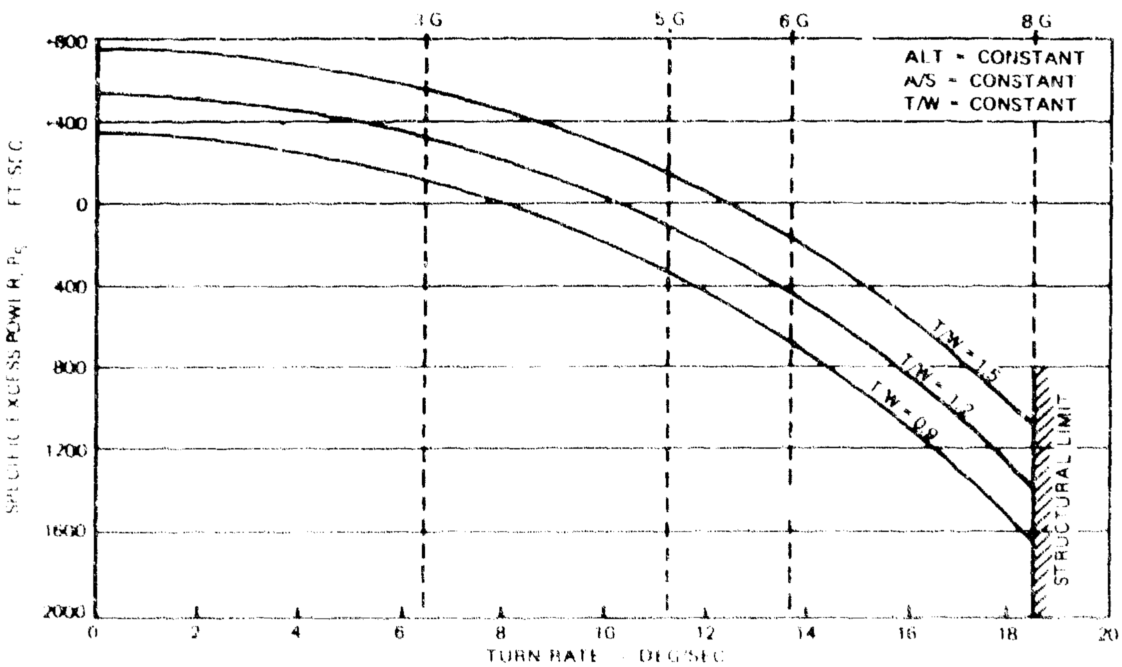

Figure 13
Effect of Improved Thrust-to-Weight

Maneuvering Performance Superiority

A conceptual evaluation of the relative maneuvering performance of two aircraft can be made by comparing their energy rate variations with turn rate characteristics as indicated in Figure 14. The friendly aircraft depicted has a margin in turn rate regardless of the energy rate and/or turn rate employed by the threat aircraft. The specific excess power margin at zero turn rate provides the friendly aircraft with the advantage of engaging or disengaging at will. The sustained turn rate margin enables the friendly aircraft to maintain an energy level above that of the threat aircraft. The maximum instantaneous turn rate and deceleration margins allow turning superiority for the friendly aircraft in defensive maneuvering situations.

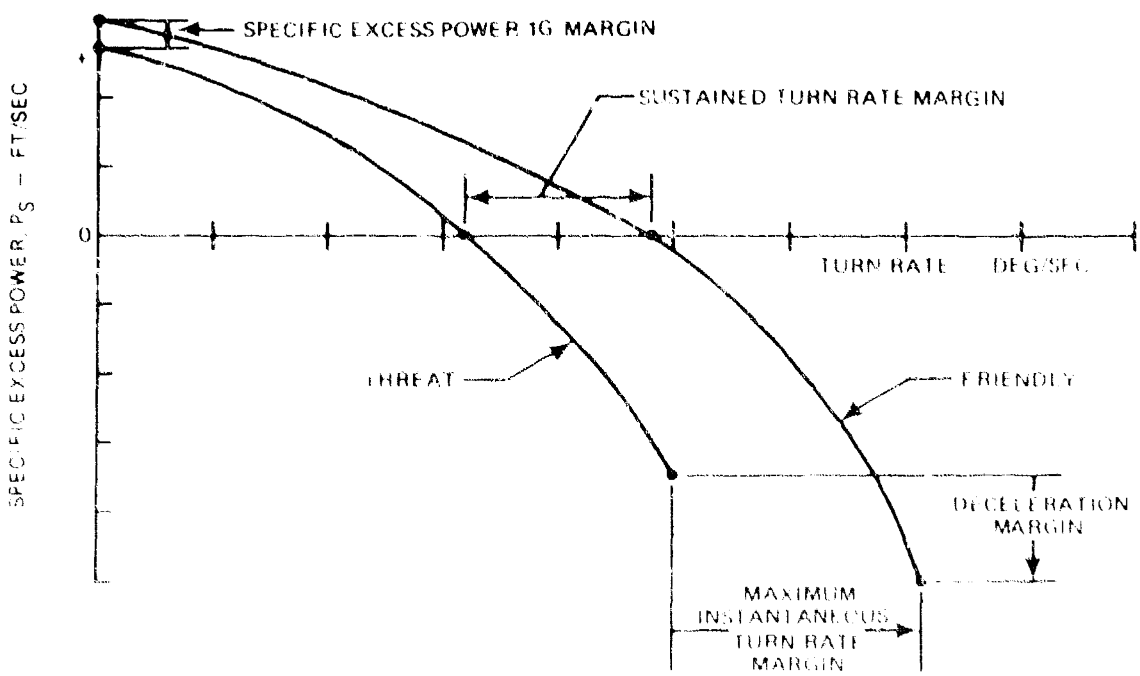

Figure 14
Energy Maneuverability Comparison

In summary, this analysis of aircraft maneuverability, particularly when supplemented by a specific excess power comparison, can be an extremely useful tool to the tactician seeking to determine flight envelopes of advantage or disadvantage. His ability to translate the results of an energy maneuverability analysis into tactical maneuvers will contribute to mission success. Of course, the ultimate measure of a pilot's achievement is still how well he takes advantage of his aircraft's capabilities, whatever they may be.

ANALYSIS OF SUSTAINED VS. INSTANTANEOUS CAPABILITIES

The major factors influencing sustained and instantaneous turning performance have been discussed. A combined graph such as Figure 10 cannot be constructed without P_s data, but much can be learned from analysis of a graph like Figure 15. A deficiency in sustained turning capability must be due to low T/W ratio and/or C_L/C_D characteristics, while a deficiency in (limit) instantaneous n_z available must be caused by low values of $C_{L_{max}}$ and/or excessive wing loading. The onset of buffet may occur at low n_z, indicating airflow separation. In the example shown: (1) a low T/W ratio may limit sustained capabilities, (2) early onset of buffet may cause degraded flying qualities and high energy losses during maneuvering flight, while (3) the aerodynamic and structural n_z limits may provide adequate turn rates and radii for all required maneuvers. A contractor might correct the deficiencies by using maneuvering leading edge slats to improve C_L/C_D and delay airstream separation (onset buffet), and by increasing engine thrust to improve sustained capabilities.

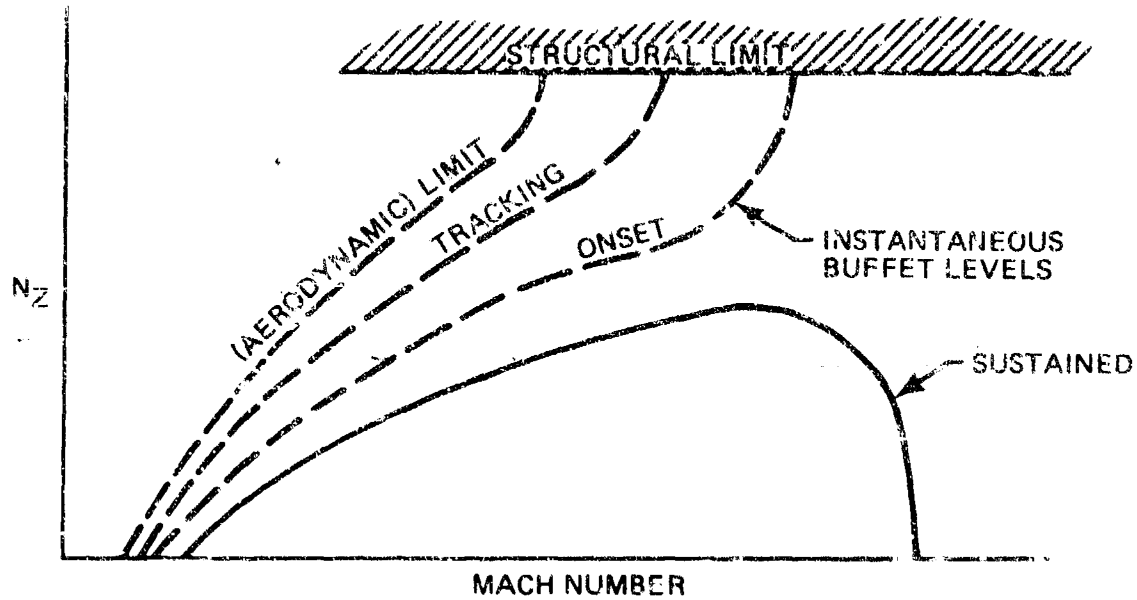

Figure 15
Engine-Airframe Compatibility

LEVEL FLIGHT TURNING PERFORMANCE

TEST PROCEDURES AND TECHNIQUES

Preflight Procedures

a. Prepare pilot's data card. In flight, in each stabilized turn, record the following cockpit data.

Airspeed.

Pressure altitude.

Normal acceleration.

Fuel used or fuel remaining.

OAT.

b. On the data card make a brief table of the variation of normal acceleration with bank angle and have a blank plot of n_z vs. V (or M) to plot rough data as collected.

Flight Procedures

a. Obtain V_{mrt} point first. Record the actual reading of the sensitive accelerometer at this 1g point. Any disparity from 1.0g should be noted as n_z tare.

b. Set a bank angle for the next data point. A 30° bank angle is recommended. Assume that stable equilibrium conditions can be obtained and use the constant altitude test technique. Maintain the 30° bank angle, test altitude, and MIL thrust until equilibrium airspeed is attained. Trim out stick forces as speed decreases. Record data when equilibrium conditions are achieved for 5 sec.

c. The constant altitude technique may be repeated for greater bank angles and higher normal acceleration until the maximum level flight, normal accelerations has been achieved. At bank angles greater than about 60°, the high variation of n_z with bank angle may make stabilization very difficult.

d. If desired, the constant airspeed technique may be used for these difficult test points. For airspeeds less than that for peak normal acceleration, the test technique will be that for unstable equilibrium conditions. The constant airspeed technique must be used. Fly smoothly and make small corrections as shown during the performance demo flight until equilibrium conditions have been achieved for at least 5 sec, then record data. Satisfactory data may be obtained within 1,000 ft of the desired test altitude.

e. Some airplanes (mainly propeller driven) exhibit different turning characteristics in left and right turns. The direction of turn should be alternated to establish whether the characteristics vary.

DATA REDUCTION AND PRESENTATION

General

The variation of maximum normal acceleration in level flight with Mach number will be presented for a standard airplane gross weight. From this characteristic (faired test data), the variation of radius of turn and rate of turn with Mach number will be computed.

Computer

Usually, the appropriate Hewlett-Packard program will be used for data reduction. This program corrects the data to standard day temperature and standard gross weight conditions. Review necessary input parameters during preflight planning.

Manual

a. Correct observed airspeed for instrument and position error.

b. Determine Mach number from the $M - V_c - H_p$ chart.

c. Correct observed normal acceleration for instrument error and for n_z tare. Assume the tare correction is constant throughout the range of the sensitive accelerometer and apply it to all observed values.

$$n = n_o + n_{ic} + n_z \text{ tare}$$

d. Compute airplane gross weight, W.

e. Compute normal acceleration for the standard weight.

$$n_s = n_t \left(\frac{W_t}{W_s}\right)$$

f. Construct a graph of the variation of standard weight normal acceleration with Mach number or velocity. From the faired curve, read values of n_s at various airspeeds.

g. Correct observed OAT for instrument error to determine T_a (if necessary).

h. Compute true airspeed

$$V_T = 65.8 \; M \; \sqrt{T_a} \; (^{o}K) \qquad \text{fps}$$

i. Calculate radius of turn

$$R = \frac{V_T^{\,2}}{g\sqrt{n_s^{\,2} - 1}} \qquad \text{ft}$$

j. Calculate rate of turn

$$\omega = \frac{57.3 \; V_T}{R} = \frac{57.3 \; g \sqrt{n_s^{\,2} - 1}}{V_T} \qquad \text{deg/sec}$$

k. Present the following graphs for each altitude tested:

Variation of sustained normal acceleration with Mach number/KCAS.

Variation of sustained radius of turn with Mach number/KCAS.

Variation of sustained rate of turn with Mach number/KCAS.

BUFFET BOUNDARY

TEST PROCEDURES AND TECHNIQUES

Preflight Procedure

a. Review stall/spin characteristics and recovery techniques. Prepare pilot's data card. For each altitude at several speeds, record the following cockpit data:

Mach number.

Pressure altitude.

Normal acceleration at onset buffet.

Normal acceleration at tracking buffet.

Normal acceleration at limit buffet.

Fuel used or fuel remaining.

AOA, remarks, etc.

Flight Procedure

a. The maneuver best suited to obtain the data required for this test is the wind-up turn in which normal acceleration is steadily increased while maintaining a constant Mach number. Stabilize at the desired airspeed with a mission-related power setting (climb may be necessary) and slightly above the assigned altitude. Smoothly roll into a turn and steadily increase n_z. When the Mach number shows a tendency to decrease, use more bank angle to increase nose down attitude. Note the normal accelerations when onset buffet and tracking buffet occur. Continue to increase n_z until limit buffet or the structural limit of the airplane is attained. Recover and record data. Satisfactory data may be obtained at any altitude within 1,000 ft of the assigned altitude. If possible, record altitude for each buffet level data point. This will provide the most accurate data and minimize altitude effects.

b. Repeat this procedure for other airspeeds until the buffet boundary for the entire maneuvering airspeed range is defined. For some Mach numbers it may be necessary to repeat the wind-up turn, obtaining onset and tracking buffet in the first turn, and limit buffet in the second. The use of augmented thrust may be useful in some airplanes to maintain speed or to reduce altitude loss during the maneuvers, but should be recorded and noted on the graphical data plot.

DATA REDUCTION AND PRESENTATION

General

The Hewlett-Packard programs will usually be used for reducing these data. Output is in the form of lift coefficient (C_L) for the various buffet levels vs. Mach number. Once the variation of C_L with Mach number is smoothly faired for each buffet level, n_z vs. M curves can be constructed using the formula

$$n_z \text{ inst} = \frac{C_L \, \rho V_T^2 \, S}{2W}$$

where S is the wing area in (ft^2). This procedure may indicate that C_L curves generalize between altitudes, which would allow generation of many curves after testing only two or three altitudes.

Manual

a. Correct observed Mach number for instrument and position error.

b. Correct observed normal acceleration for instrument error, and for n_z tare.

c. Compute airplane gross weight.

d. Compute normal acceleration for a standard weight.

$$n_s = n \left(\frac{W}{W_s}\right)$$

e. Construct graphs of the variation of normal acceleration for onset, tracking, and limit buffet with Mach number for each test altitude.

f. Compute and construct graphs of:

Variation of instantaneous rate of turn with Mach number/KCAS.

Variation of instantaneous radius of turn with Mach number/KCAS.

SECTION IX

CLIMB PERFORMANCE

REFERENCES

SECTION IX

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CLIMB PERFORMANCE

PURPOSE

The purpose of this test is: (1) to determine the climb performance of a turbojet airplane; (2) to determine compliance with the combat ceiling (500 fpm rate of climb) or climb performance guarantee of the applicable Detail Specification; and (3) to check proposed climb schedules and obtain data for presentation in Flight Handbooks.

DISCUSSION AND THEORY

The excess power (specific energy) characteristics of an airplane can be determined at several altitudes using the level acceleration run tests described in Section VI. Maximum rate of climb and maximum energy climb schedules can be derived from these data using the procedures set forth in Section XII. Since computation of time to climb data from acceleration runs (or sawtooth climbs) is tedious and does not always yield accurate results, time to climb data are normally determined by flying continuous climbs to service ceiling.

Normally, the contract guarantee ceiling would be checked using the contractor's recommended climb schedule. For the purposes of this evaluation, TPS derived schedules will be flown. The maximum energy climb schedule results in minimum time to a given energy altitude. Variations of this schedule may result in minimum time to a pressure altitude, maximum range or greater cruise ceiling. Close approximation may be obtained by using a constant KCAS/Mach schedule which is easier for a pilot to fly.

An important aspect of climb performance is the time and fuel required to accelerate from brake release to the climb schedule airspeed near sea level. Test climbs are usually divided into two phases: (1) the acceleration and transition to climb schedule and (2) the climb from near sea level to the ceiling.

In performing the climb test, it is important to fly the desired climb schedule precisely and smoothly. If the pilot strays from the schedule momentarily but corrects his airspeed smoothly, the average rate of climb for the interval will closely approximate the rate of climb which would have been achieved had the schedule been flown precisely. This is based on the principle of interchangeability of potential and kinetic energy and was applied in the acceleration runs test also. There are energy losses in this exchange, of course, and the effect of deviations from the desired climb schedule becomes more critical as altitude increases.

Another factor which must be considered in the climb test is the effect of wind velocity gradient with altitude. If the climb were flown at constant true airspeed into a headwind which increased with altitude, the velocity of the airplane (relative to the ground) would decrease as altitude increased. Some of the airplane's kinetic energy would be converted to potential energy (in effect, a zoom maneuver), and the observed rate of climb would be greater than that observed on a similar climb in still air conditions. Correction for horizontal wind gradients with altitude and for vertical movements of the air mass are tedious and beyond the scope of this exercise. Gradient effects can be minimized by performing the test climbs on a heading perpendicular to the average winds.

The correction of jet climb performance data to standard day conditions is similar to the corrections required for acceleration run data - temperature correction for thrust, momentum correction for weight, and induced drag correction. The thrust correction usually is the most significant although the weight and induced drag corrections must always be considered. An assumption in the correction of climb data is that the standard day calibrated airspeed at a given altitude is the same as the test day airspeed. When evaluating a climb performance guarantee which prescribes a standard rate of change of airspeed, an additional correction to rate of climb for nonstandard dV/dh or dV/dt must be made.

TEST PROCEDURES AND TECHNIQUES

Preflight Procedures

a. Required data (Photopanel required - set on 1/2 frame/sec).

Time.

Pressure altitude.

Airspeed.

OAT.

Fuel used (or remaining).

RPM (for reference).

Take photopanel data every 4,000 ft interval for altitudes below 30,000 ft and every 2,000 ft for higher altitudes but no less frequently than every 2 min. For high performance airplanes, it is usually desirable to take continuous photopanel data to a medium altitude. Take correlating data on kneeboard.

b. Prepare data card (see Figure 1).

c. Obtain assigned climb schedule from group leader.

d. Obtain upper air wind velocities from aerology. Estimate average direction weighting strong vertical gradients more heavily.

e. Review takeoff technique.

f. Carry a 60-sec stopwatch or navigator's hack watch.

WEATHER

CARD NO. QUK-1

CLIMB TO CRUISE CEILING

AIRPLANE TYPE	BU. NO.	TIME T.O. / TIME LAND	DATE
PBY-4	001394		

PILOT	T.O.C.G. / GEAR DOWN	%
KIDD		24.5 %

CONDITION	T.O. GROSS WEIGHT
TRAINER	51,346 lb

EXTERNAL CONFIGURATION

2 OAT PROBES

TO: TIME ____ SEC DISTANCE ____ ft

Hpa	Vo	Δt	P/P	CN			REMARKS
FIELD ELEV	PARTICULAR CLIMB SCHEDULE	0					
ALT FOR INTERRUPTION		—					
3000		0					
7000							
11,000							
↓							
CRUISE CEILING							

Figure 1
Sample Pilot's Data Card

Flight Procedures

The climb performance test will be conducted in two parts - brake release to 3,000 ft (an arbitrary altitude) and climb to cruise ceiling from near sea level. Estimated takeoff performance will also be determined during the first part of the test.

a. Energize the photopanel and release brakes at a predetermined film counter number. The takeoff should commence at a runway marker so that a reasonable estimate of test day takeoff performance can be made. Record time and distance to lift-off. Start watch at brake release. Record time and fuel required for takeoff, acceleration, and climb to 3,000 ft above field elevation. The acceleration phase should be conducted at as low an altitude as possible commensurate with safety, course rules, and sound judgment.

b. Establish the airplane on a heading perpendicular to the average wind direction. Energize photopanel. Set MIL thrust and stabilize on the required climb schedule to pass through 2,500 ft pressure altitude in a stabilized, trimmed climb. When the takeoff heading is within 30 deg of the climb heading, the climb may be done in one continuous maneuver from brake release. Do not exceed 10 deg bank angle while adjusting heading in the climb.

c. Commence timing the climb at 3,000 ft pressure altitude. Record kneeboard and photopanel data at 4,000 ft intervals. Let the photopanel run continuously until well established in the climb. Thereafter, energize the photopanel for 10 to 20 sec while passing through each data altitude.

d. Above 30,000 ft record data at 2,000 ft intervals, but no less frequently than each 2 min until cruise ceiling is attained. Determine that the rate of climb is less than 300 fpm by observing a time history of altitude. It is important that this part of the climb schedule be flown precisely.

e. If it is necessary to reduce thrust to maintain EGT limit, note this on the data card.

f. Should it become necessary to interrupt the climb for any reason, the gross weight change must be kept to a minimum. At low altitude, this is best accomplished by a rapid descending turn at idle thrust supplemented by the use of speedbrakes, if necessary. At high altitudes, it may be necessary to make the turn at MIL or MAX thrust to prevent excessive altitude/airspeed loss and suffering the attendant increase in fuel used attempting to regain lost altitude and/or airspeed.

g. The airspeed indicator, vice the Machmeter, should be used for the climb schedule since the Machmeter is of insufficient accuracy.

MANUAL DATA REDUCTION

General

The climb data may be presented for test day conditions. For a more exact analysis, the data should be corrected for the effects of nonstandard thrust and gross weight. Record the following data from the photopanel film -- at brake release, at climb airspeed and at 3,000 ft above field elevation; in the climb at 4,000 ft intervals from 3,000 ft pressure altitude to 30,000 ft and at 2,000 ft intervals above 30,000 ft, but at least every 2 min.

Elapsed Time H_p V W_F OAT N

Data Reduction - Test Day Climb Performance to Cruise Ceiling

a. Correct observed altitude and airspeed for instrument and position errors.

b. Determine Mach number from $V_c - M - H_p$ chart.

c. Correct observed OAT for instrument error and determine the ambient temperature from Appendix IV.

d. Calculate true airspeed:

$$V_T = 39.0\, M \sqrt{T_a} \quad (°K) \qquad \text{kt}$$

e. Compute fuel used in lb.

f. On a sheet of 16" x 20" graph paper, construct a plot of the variation of pressure altitude with the time for both parts of the climb test. Use scales of about 5,000 ft/in. and 5 min/in. Assume that the data recorded at field elevation and 3,000 ft above field elevation are equivalent to data which might have been recorded on another day at sea level and 3,000 ft pressure altitude. On the same graph sheet also plot the variation with altitude of calibrated airspeed or Mach number (depending on climb schedule used), both test day and standard day ambient temperature and fuel used. Fair a smooth curve through the time to climb and airspeed data. In fairing these curves, observe that when the airspeed was faster than the mean, the climb data points will fall below the faired climb curve and vice versa. There may be a slight discontinuity in the time to climb curve at an altitude when it was necessary to reduce thrust, or at an altitude where the temperature lapse rate changed abruptly. Fair a curve through the fuel used data. Extrapolate both the time to climb and fuel used curves to sea level.

g. From the faired time to climb curve, graphically measure the indicated rate of climb, $(dH_p/dt)_t$ at 5,000 ft intervals beginning at sea level. Plot the variation of indicated rate of climb with altitude. Mark the combat ceiling (500 fpm) and cruise ceiling (300 fpm).

h. From the faired curves of time and fuel used, record the time to climb and fuel required at 5,000 ft intervals for a climb from sea level (at climb airspeed) to cruise ceiling.

i. Compute the time to climb through each 5,000 ft interval, Δt.

j. For each interval, compute the no-wind distance flown.

$$\Delta d = V_{T_{av}} \frac{\Delta t}{60} \qquad \text{nmi}$$

k. Compute the distance flown from start of climb to each altitude.

$$\Delta d_H = \sum_{SL}^{H} \Delta d \qquad \text{nmi}$$

l. Determine the time and fuel required for takeoff and acceleration to climb airspeed by subtracting time and fuel required for climb from sea level to 3,000 ft (extrapolation of climb data) from the observed time and fuel required from brake release to 3,000 ft.

m. These results may be presented as the table of time, fuel used, and air distance (from sea level at climb airspeed) to cruise ceiling in 5,000 ft intervals of altitude. State the time and fuel required for takeoff and acceleration in a footnote to the table.

Data Reduction - Ceiling Guarantee

The rate of climb observed at some arbitrarily chosen altitude can be corrected to standard conditions of thrust and gross weight. If such corrections are made to rate of climb observed at several altitudes near the guarantee ceiling, a curve defining the variation of standard day rate of climb with altitude may be drawn, and the guarantee ceiling at standard conditions of thrust and gross weight established. The ceiling also is affected by airspeed. This test was flown at several climb schedules, one of which will result in the greatest ceiling for the conditions of the guarantee.

n. For the several data points obtained at 34,000 ft and above correct observed pressure altitude and airspeed for instrument and position errors.

o. Determine Mach number from V_c - M - H_p chart.

p. Correct observed OAT for instrument error and determine ambient temperature from Appendix IV (note - some of the ambient temperature data computed in step c may be used if suitable)

q. Calculate true airspeed.

$$V_T = 65.8\ M \sqrt{T_a}\ (°K) \qquad \text{fps}$$

r. Determine airplane gross weight from fuel used or fuel remaining.

s. Obtain pressure ratio, δ, and standard day ambient temperature T_{a_s}, from Appendix VII.

t. Plot the variation of time to climb and true airspeed with altitude. Fair a curve through these data. Graphically determine indicated rate of climb, $(dH_p/dt)_t$, at 36,000 ft for at least three higher data points.

u. Calculate actual rate of climb for test day.

$$(R/C)_t = \left(\frac{dH_p}{dt}\right)_t \left(\frac{T_a}{T_{a_s}}\right) \qquad \text{fpm}$$

v. Compute net thrust correction (nonstandard temperature).

$$\Delta(R/C)_1 = \left(\frac{60V_T}{W_s}\right) \left(\frac{dF_n/}{dT}\right) (T_{a_s} - T_a) \qquad \text{fpm}$$

where

$$\frac{dF_n/\delta}{dT} = -70 \text{ lb/}^{\circ}\text{K} \quad \text{(J57-P-4/8/16 engine)}$$

$$= -40 \text{ lb/}^{\circ}\text{K} \quad \text{(J-52-P8A engine)}$$

$$= -46 \text{ lb/}^{\circ}\text{K} \quad \text{(J-85 engine)}$$

w. Compute induced drag correction.

$$\Delta(R/C)_2 = \left(\frac{120gR}{P_{s_1}\, eb^2 W_s}\right) \frac{T_{a_s}}{V_T} (W^2 - W_s^{\,2}) \qquad \text{fpm}$$

where

$$\frac{120gR}{P_{s_1}\, eb^2 W_s} = 5.4 \times 10^{-6} \text{ for A-4}$$

$$= 10.8 \times 10^{-6} \text{ for T-38}$$

$$= 1.05 \times 10^{-6} \text{ for F-8K}$$

x. Compute standard day rate of climb for the conditions of the guarantee.

$$(R/C)_s = \frac{W}{W_s} (R/C)_t \left[\frac{T_{a_s}}{T_a}\right]^{1/2} + \Delta(R/C)_1 + \Delta(R/C)_2$$

y. Plot the variation of standard rate of climb with altitude. Extrapolate as necessary to determine service ceiling. On the same graph, show the test day rate of climb variation and the test conditions (average gross weight and average difference of ambient temperature from standard).

COMPUTER DATA REDUCTION

Climb data is normally reduced by CSD (Computer Services Division) with resultant plots reduced to standard day conditions. Determination of climb schedules from acceleration run data is contained in Section XII of this manual.

SECTION X

DESCENT PERFORMANCE

REFERENCES

Section X

1. Petersen, Aircraft and Engine Performance, Chapter 10.

2. USAF, ARPS PTM, Section 4.7 of Theory.

DESCENT PERFORMANCE

PURPOSE

The purpose of this test is to determine the idle thrust descent performance - time, fuel used, and range - in a descent from 40,000 ft to near sea level at landing gross weight. Such descent tests are required to determine the most desirable airspeed for descent and to obtain data for presentation in Flight Handbooks.

DISCUSSION AND THEORY

For stabilized, level flight, engine thrust is adjusted to balance the airplane drag. In terms of power, we say that engine power equals power required (FV = DV, ft-lb/sec). If the thrust were reduced to zero, then the power required to maintain the airplane's airspeed must come from the airplane's time rate of change of kinetic and potential energy. The rate of expenditure of this energy varies directly with the rate of descent and linear acceleration. The minimum rate of descent will occur at the airspeed for minimum power required. The maximum gliding range will be obtained at a faster airspeed where the maximum ratio of true airspeed to power required occurs.

The airspeed for minimum rate of descent at idle thrust in the jet airplane is approximately the airspeed which results in the maximum ratio of $C_L^{3/2}/C_D$. The airspeed for maximum range in the idle thrust glide is approximately the airspeed for maximum ratio of C_L/C_D. The idle thrust descent airspeed schedule can be estimated on the basis of these theoretical ratios; however, for a more precise result, the effects of residual thrust also must be considered. The estimated idle descent airspeed schedule can be refined by performing a series of descents at several constant airspeeds. Usually, a constant calibrated airspeed descent

schedule which will give near optimum performance can be determined. For this test, idle thrust descents will be made at one gross weight, and the effects of weight changes will not be considered. Qualitatively, an increase in airplane gross weight has the effect of increasing the airspeed required to maintain the desired lift coefficient which is stated above. The rate of descent observed at the heavier gross weight will be greater because the power required is greater; however, the glide angle will not change, and the range in the descent will be unchanged by an increase in weight. This statement must be qualified in that it does not consider the effect of residual thrust at the idle thrust setting or steady winds. Generally, the residual thrust effects result in increased descent range as airplane weight decreases. Because a change in airplane kinetic and potential energy is involved in the idle thrust descent test, wind gradients will affect the observed rate of descent. The descent must be flown on a heading perpendicular to the average wind direction to minimize vertical wind gradient effects.

TEST PROCEDURES AND TECHNIQUES

Preflight Procedures

a. Required Data (photopanel desired - 1/2 frame/sec)

Time

Pressure altitude

Airspeed

OAT

Fuel used

RPM (for reference only)

Record both photopanel and kneeboard data at 5,000 feet intervals commencing at 40,000 ft. Record last data at 3,000 ft.

b. Prepare data card (see Figure 1).

c. Obtain upper air wind velocities from aerology. Estimate average wind direction, weighting strong vertical gradients more heavily.

d. Carry a 60-sec stopwatch or navigator's hack watch.

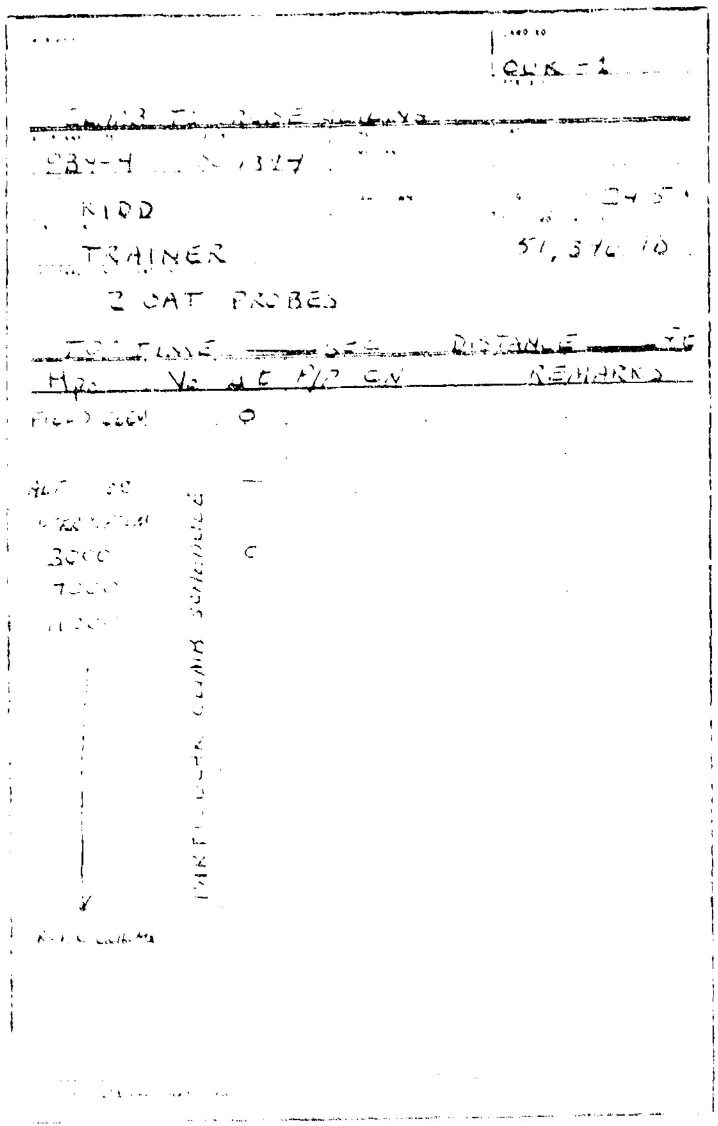


Figure 1
Sample Pilot's Data Card

Flight Procedures

Normally, the idle thrust descent test will be performed in the same flight as the service ceiling climb test. It is desirable to perform the descent along a reciprocal flight path of that used for the climb because it will probably allow the ambient temperature information obtained in one of the tests to be used for both.

a. Proceed to the required starting point at about 42,000 ft. When fuel remaining is 2,000 lb (1,600 lb in T-38A) establish a heading inbound to the field and perpendicular to the average wind direction. Stabilize in level flight at the required airspeed. Set idle thrust and quickly retrim so as to be stabilized in the glide at 40,000 ft.

b. Energize the photopanel briefly and record required data at 40,000 ft and at 5,000 ft intervals in the descent. Record last data at 3,000 ft pressure altitude.

c. At 3,000 ft after mentally noting required data and advancing the throttle, record the data on the kneeboard. Be alert to the possibility of fuel control icing and be prepared to take proper corrective action immediately if the engine fails to accelerate.

MANUAL DATA REDUCTION

General

The descent data will be presented for test day conditions. For a more exact analysis, the rate of descent should be corrected to a standard gross weight. Corrections to thrust are not significant in this test. Record the following data from the photopanel film at 5,000 ft intervals beginning at 40,000 ft and also for 3,000 ft:

Elapsed time	H_p	V	W_F	OAT	w_f*	N*

*for reference only

Data Reduction - Idle Thrust Descent

a. Correct observed altitude and airspeed for instrument and position errors.

b. Determine Mach number from V_c - M - H_p chart.

c. If the idle descent was performed over the same flight path as the service ceiling climb and the weather conditions were stable, use ambient temperature calculated in step c of the climb data reduction. Otherwise, correct observed OAT for instrument error and determine the ambient temperature from Appendix IV.

d. Calculate true airspeed

$$V_T = 39.0 \, M \sqrt{T_a} \; (°K) \qquad \text{kt}$$

e. Compute fuel used or fuel remaining in pounds.

f. On a sheet of 16" x 20" graph paper, construct a graph of the variation of pressure altitude with time in the descent. Use scales of about 5,000 ft/in and 5 min/in. Fair a smooth curve through the data and extrapolate to sea level. On the time to descent curve, note the CAS schedule flown.

g. From the faired time to descent curve, graphically measure indicated rate of descent, $(dH_p/dt)_t$ at 5,000 ft intervals beginning at 40,000 ft. Also plot this data on the large graph.

h. From the faired curves of time and fuel used, record the time to descend and the fuel used in a descent from 40,000 ft to sea level. Repeat at 5,000 ft intervals.

i. Compute time to descend through each 5,000 ft interval, Δt.

j. For each interval, compute the air distance flown.

$$\Delta d = V_{T_{av}} \frac{\Delta t}{60} \qquad \text{nmi}$$

k. Compute the glide distance from each 5,000 ft interval to sea level (note that in steps h and k that the reference point is the terminus of the glide - sea level).

$$\Delta d_H = \sum_{S\,L} d \qquad \text{nmi}$$

l. The results may be presented as a table of time, fuel used, and air distance for an idle thrust descent to sea level from 5,000 ft intervals of altitude (to 40,000 ft).

m. Note - had fuel flow information been desired, more reliable data are obtained from differentiation of the variation of fuel used with time than from the fuel flow indicator, which generally is inaccurate at low flow rates.

$$w_{f_{av}} = \frac{W_{F_b} - W_{F_a}}{\Delta t} \qquad \text{lb}$$

COMPUTER DATA REDUCTION

See Section XIII.

SECTION XI

STOL PERFORMANCE

NOTATIONS INTRODUCED IN THIS SECTION

R/C	Rate of Climb
γ	Flight path angle
$V_{\gamma_{max}}$	Airspeed for greatest flight path climb angle
$V_{R/C_{max}}$	Airspeed for greatest rate of climb
Q	Engine torque pressure
N_1	Engine speed - percent

INVESTIGATION OF STOL PERFORMANCE

PURPOSE

The purpose of this test is to determine the STOL performance of a fixed incidence turboprop airplane. Airspeed for stall, best climb angle, best rate of climb, and best approach angle will be determined. Test day takeoff and landing distances will be determined.

DISCUSSION AND THEORY

The performance of a STOL airplane will depend greatly upon its engine thrust line angle, its variable geometry or incidence features, and its gross weight. In the case of the fixed incidence STOL airplanes, the principle source of lift will be the wing, therefore, the excess power characteristics will dictate its potential takeoff and landing performance. Additional limitations may be imposed by flying quality deficiencies.

The sawtooth climb method can be used to determine excess power characteristics and airspeeds for best angle and rate of climb. The best approach configuration and airspeed can be determined by a sawtooth descent method. The effect of gross weight on these airspeeds cannot always be calculated by the classical square root weight ratio method since the airplane pitch angle may be so large that the thrust component supplies a significant amount of lift. This means that test data points must be taken at both heavy and light gross weights. Relatively small variations in gross weight can be corrected by the classical method in order to correct test data to one weight. After best airspeeds have been determined, actual takeoff and landing distances must be measured. A number of special test recording devices are available for takeoff and landing data collection

and analysis. The Fairchild flight analyzer is a typical device used for determining takeoff and landing performance. The touchdown rate of descent indicator (TRODI) is a typical carrier suitability test device which can be used to rapidly determine sink rate.

Special pilot techniques may be required for STOL operation and performance testing. An airplane with a fixed incidence wing and a fixed thrust axis will require particular pilot attention to ensure that the angle of attack for maximum lift is established at the earliest possible point in the takeoff roll. If sufficient longitudinal control is available, simply holding full aft stick will achieve the desired angle of attack before minimum lift-off speed is attained. If either the elevator control power is marginal or the location of the main landing gear precludes the airplane's achieving the necessary angle of attack prior to attaining minimum lift-off speed, it may be possible to "bounce" the airplane to the desired attitude. This is done by compressing the nose strut with a full forward stick input followed by a full aft stick input. This technique can result in attaining the desired airplane attitude at slower airspeeds than from static control inputs. The airplane will usually become airborne with full aft stick. After becoming airborne, it may be necessary to ease the stick forward slowly to accelerate to the airspeed for best climb angle, or it may be possible to simply hold full aft stick while the airplane climbs to some desired altitude without accelerating to a higher airspeed.

Unfortunately, the takeoff airspeed and possibly the airspeed for best climb angle will usually be less than the minimum single engine control airspeed. The pilot must always be prepared to abort a takeoff at the first sign of significant loss of power from an engine. This possibility must be considered when recommending a landing gear up or down climb configuration.

The importance of best climb angle for clearing an obstacle after takeoff is obvious. The airspeed for best climb angle will be lower than the airspeed for best rate of climb. While there will be a best climb airspeed for every airplane, the method of accelerating to that speed, or the requirement for actually accelerating to that airspeed will depend upon the airplane's performance and mission.

The short field landing also requires some special pilot techniques. Minimum landing distance will depend upon a combination of reverse pitch and brakes. Reverse pitch is usually the primary means of controlling landing roll distance. As a general rule, if full power is available in reverse pitch, little or no braking will be required. When less than 50% power is available in reverse pitch, brakes will be required, and special pilot precautions must be taken to avoid blown tires. Whenever any amount of reverse pitch is used, it must be applied while the engine power is still relatively high. This means the approach should be made with the highest possible power requirement. This may result in airspeeds that are on the unstable side of the power required curve. Again, this airspeed will usually be lower than the minimum single engine airspeed. The approach may be a constant rate of descent (mirror type) or a level approach with a fixed throttle high dip and flare to touchdown (flap paddles type). The type of approach will be dictated by the requirement to maintain sufficiently high power for effective reverse pitch operation and a sufficiently safe airspeed-power-altitude relationship to avoid single engine catastrophe. As soon as the airplane touches down, full available reverse power must be applied. If nose wheel steering is available, the nose wheel must be held on deck during reverse pitch operation. In some short field work (C-130 carrier trials for instance), reverse pitch may be applied prior to actual touchdown. This technique usually requires an outside observer or LSO, and will not be used at USNTPS.

TEST PROCEDURES AND TECHNIQUES

Pertinent Particulars

a. At the time these tests are performed, the student test pilot should have completed both the performance and the longitudinal, lateral-directional syllabus test flights.

b. Determine stall characteristics before conducting low speed tests near the ground.

c. Ensure that autofeather feature operates properly prior to conducting actual takeoff and landing tests.

Preflight Procedures

a. First flight required data.

1. Stall airspeed at heavy and light weights.

2. Sawtooth climb information, V_o, R/C_o, H_{p_o}, W_R, Configuration, and Remarks.

3. Sawtooth descent information, V_o, R/C_o, H_p, W_R, Configuration, Torque, RPM, N_1.

b. Second flight required data.

1. Takeoff data. Configuration, $V_{o_{NLO}}$, $V_{o_{TO}}$, t_{TO}, t, t_{NLO}, H_p, S_{TO}.

2. Landing data. $V_{o_{PA}}$, V_{o_L}, V_{o_R}, t, S_L, Configuration Q_{PA}, RPM_{PA}.

3. Prepare pilot's data cards (see Figures 3, 4, and 5).

Flight Procedures

First Flight:

Determine stall airspeeds, airspeeds for best climb angle and rate of climb, and best approach airspeed and configuration for approach. The airspeed range for this investigation should be from V_{stall} to a midcruise airspeed. Configuration takeoff should be investigated with LG down and retracted. Power approach configuration should be investigated with speedbrakes in and out, full flaps, gear down. At least two constant power settings should be investigated. The most desirable approach configuration will be one that requires sufficient power to ensure that engine RPM is high, but does not require a power setting that would preclude successful recovery from a sudden engine failure. Data must be obtained at a relatively heavy and a relatively light weight.

Sawtooth climbs and descents will use indicated rate of climb or descent in lieu of a timed climb. Indicated rate of climb information will be sufficiently accurate for STOL evaluation purposes. If refinement of results is required, the time method could be employed.

The recommended sequence of events on the first flight is as follows: After a normal takeoff, climb to 7,000 ft H_p and determine stall airspeeds in each configuration to be tested. Then descend to 3,000 H_p and conduct sawtooth climbs and descents. Start at the high airspeed and collect data by alternating climbs and descents. Take test data from base altitude of 3,000 ft H_p and ensure that data taken near stall airspeed is taken at an altitude near 7,000 ft H_p. Data will be recorded by the acting copilot when the pilot has established equilibrium conditions for at least 30 sec. Determine stall airspeeds at 7,000 ft and with approximately 400 lb of fuel remaining. Make a normal landing.

Second Flight:

The second flight will be used to determine takeoff and landing distances and proper pilot techniques for minimum STOL distances. Takeoffs should be investigated by using neutral stick and then rotating normally at lift-off airspeed by using full aft stick and by using a dynamic lift-off technique. The dynamic technique is full back stick at V_{STO-10} kt. At this airspeed, apply full forward stick followed by full aft stick. The effect of cocking the airplane to overcome initial torque should also be investigated. Nose wheel steering should be used for all takeoffs. STOL's are not to be conducted with autofeather inoperative. Takeoff time, distance, and lift-off speed will be recorded.

Landing approach and landing will be evaluated using both flat and descending approaches. Reverse pitch and nose wheel steering will be used. Brakes should be used only after the landing roll is essentially complete. Landing distance, approach airspeed, touchdown airspeed, and configuration will be recorded.

DATA REDUCTION AND PRESENTATION

First Flight

a. Correct stall data to indicated stall airspeed at one heavy and one light standard weight.

$$V_o + V_{ic} = V_i$$

$$V_{is} = V_{it} \left(\frac{W_s}{W_t}\right)$$

b. Correct sawtooth climb data to indicated airspeed and observed rate of climb at heavy standard weight for heavy weight data points, and at light standard weight for light weight data points.

$$V_o + V_{ic} = V_i$$

$$V_{is} = V_{it}\left(\frac{W_s}{W_t}\right)^{1/2}$$

$$R/C_{o_s} = R/C_{o_t}\left(\frac{W_s}{W_t}\right)^{1/2}$$

c. Plot variation of observed rate of climb with indicated airspeed, in configuration TO at both standard weights (see Figure 1).

d. Determine airspeed for best rate of climb.

e. Determine airspeed for greatest climb angle by plotting a tangent as shown in Figure 1. The airspeed at the tangent point represents best climb angle airspeed since

$$\left(\frac{R/C}{V}\right)_{max} = \sin\gamma_{max}$$

f. Determine whether there is any advantage in retracting the main landing gear to achieve best climb angle after takeoff from an analysis of sawtooth climb data. Consider operational aspects. Figure 2 is an example of results which would indicate gear should be kept down.

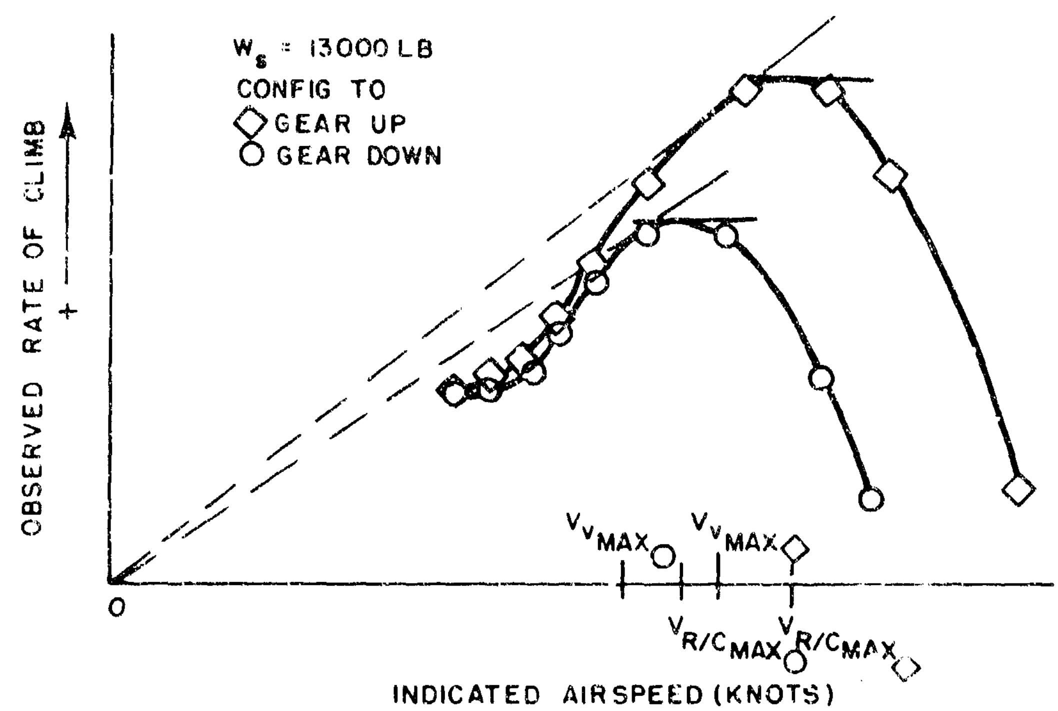
g. Correct sawtooth descent data to indicated airspeed and observed rate of descent at 11,000 lb standard gross weight and plot results as shown in Figure 1.

h. Determine airspeeds for minimum rate of descent approach and descent angle at that airspeed for both power settings investigated. See example in Figure 1.

i. Plan second flight from test results determined in steps a through h. Include evaluation of at least two airspeeds and configurations for landing evaluations.

Second Flight

j. Discuss the most successful STOL pilot techniques which you determine from these tests.



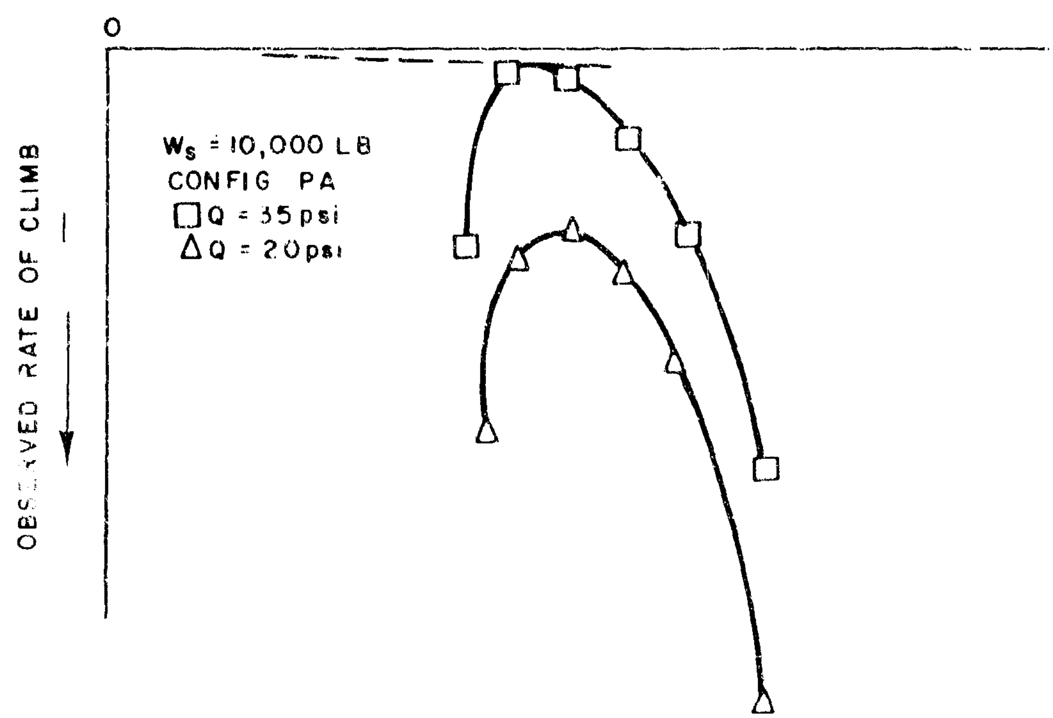


Figure 1
Sample Data Presentation

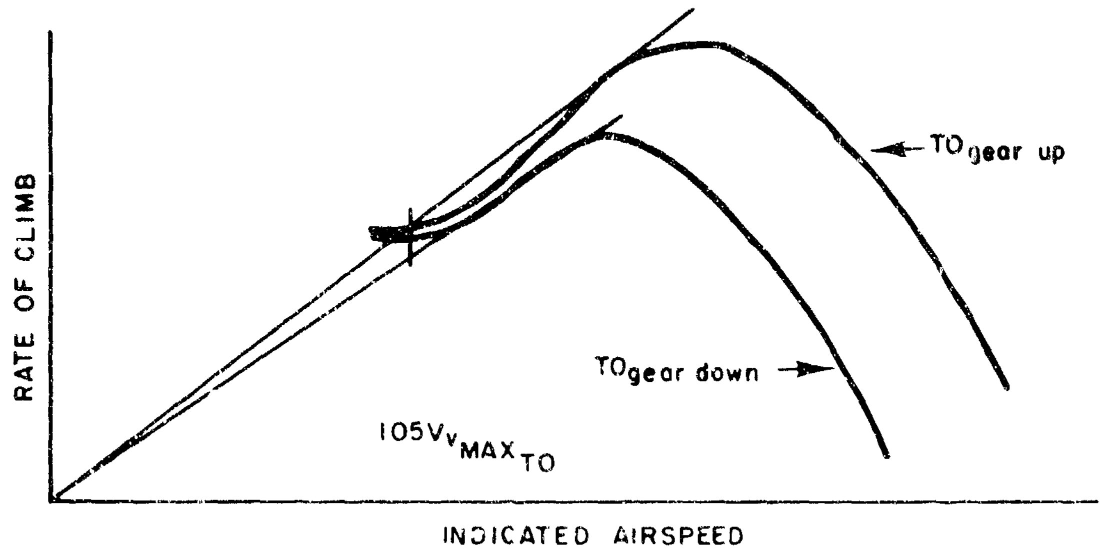


Figure 2
Sample Data Presentation

WEATHER

CARD NO.

PTR BIS

AIRPLANE TYPE	BU. NO.	TIME T.O. / TIME LAND	DATE
OV-1A	137815		9 Sept 1978

PILOT

T.O.C.G.

GEAR DOWN % UP %

CONDITION

NORMAL OBSERVATION

T.O. GROSS WEIGHT

EXTERNAL CONFIGURATION

WING TANKS

S.E Speed
TO ____ PA ____

STALL EVALUATION

Config	V_o	H_{p_o}	W_R	O_c	RPM_o	Θ_c	Remarks
CR					1450		
TO					1690		
PA					1690		
TO_1					1690		
L					1690		

SAWTOOTH CR DATA - CONFIG TO | OAT ____

V_o	140	120	110	100	95	90	85	80	75
R/C_o									
H_{p_o}									
W_R									

SAWTOOTH PA DATA - CONFIG PA

V_o	115	105	95	90	85	80	75	RPM - 1690
R/C_o								Q,
H_{p_o}								N_1 -
W_R								

FLIGHT DATA PRNC NATC 1900

Figure 3
Pilot's Kneeboard Card
First Flight

WEATHER

TAKEOFF DATA

T_a ______ WIND ______

CARD NO.

PTR 515

AIRPLANE TYPE	BU. NO.	TIME T.O. / TIME LAND	DATE
CV-1A	137815		10 Sept 1975

PILOT

T.O.C.G.

GEAR DOWN

G UP

CONDITION

NORMAL OBSERVATION

T.O. GROSS WEIGHT

EXTERNAL CONFIGURATION

CLEAR

t_1 = time to lift

s = dist to use

h = radio alt

RUNWAY ______

V_{TO}	t_{TO}	t_1	S_{TO}	S_1	Wa	Remarks

(Include comments on technique used for each takeoff)

FLIGHT DATA CARD

Figure 4
Pilot's Kneeboard Card
Takeoff Data

WEATHER

LANDING DATA

Ta ____ Wind ____

CARD NO.

PTR BIS

AIRPLANE TYPE	BU. NO.	TIME T.O. / TIME LAND	DATE
OV-1A	137815		10 Sept 1978

PILOT

T.O. C.G.

GEAR DOWN % UP %

T.O. GROSS WEIGHT

CONDITION

NORMAL OBSERVATION

EXTERNAL CONFIGURATION

CLEAN

RUNWAY ____

V_{APP}	V_{TL}	S_L	W_L	RPM	Q	N_1	Remarks

(Include comments on technique used)

FLIGHT DATA PHOTO HATC 3500

Figure 5
Pilot's Kneeboard Card
Landing Data

SECTION XII

DETERMINATION OF CLIMB SCHEDULES FROM ACCELERATION RUN DATA

DATA REDUCTION PROCEDURE FOR DETERMINING CLIMB SCHEDULES FROM ACCELERATION RUN DATA

GENERAL

The variation of rate of change of specific excess power or specific energy (P_s or dE_h/dt) at MIL thrust at several altitudes was determined from acceleration runs. To determine a climb schedule from these results, the acceleration run data must be plotted in a single graph, faired as a family, and cross-plotted as lines of constant rate of change of P_s on coordinates of altitude and airspeed.

In reducing the acceleration run data, corrections were made for nonstandard airplane weight. The corrections for gross thrust and ram drag variations because of nonstandard temperature were assumed to be negligible and were not made for manual data reduction. If automatic data reduction was employed, these corrections were included. In the preliminary fairing of manually reduced data, it may be necessary to consider the effect of nonstandard temperature.

DATA REDUCTION

In the acceleration run test, the results were presented in the form of variation of $P_s = dE_h/dt$ with Mach number or CAS for several altitudes. Further data reduction will commence with those results.

Using a 16" x 20" graph sheet, plot the variation of P_s with M for each altitude. Lightly sketch a curve through the data and label with the altitude and T_a (if reduced manually). When the curves for all test altitudes are plotted on this graph, they should form a family similar to Figure 1. It may be necessary to adjust some of the curves to fit this family. Two effects may be noted:

a. Warmer than standard ambient air temperature will tend to shift the curves downward in relation with those obtained at standard conditions. This feature is more apparent at high altitudes.

b. The curves will tend to break downward sharply at the same Mach number.

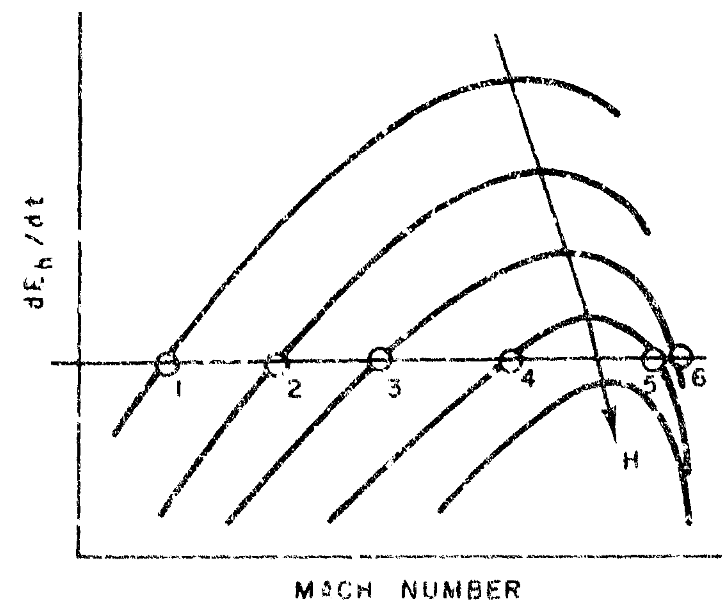


Figure I
Specific Excess Power Working Plot

When the fairing of the working plot is complete, cross-plot the data as lines of constant P_s on coordinates of altitude and true airspeed - use the prepared Specific Energy Analysis graph.

a. At several values of P_s, record the M corresponding to each altitude, (points 1 through 6 in Figure I).

b. Compute V_T for each value of M obtained.

$$V_T = 38.9 \; M \; \sqrt{\theta a_0} \qquad kt$$

c. Plot these data as lines of constant P_s on the Specific Energy Analysis graph. This plot should form a family similar to Figure 2. If unusual bends or anomalies occur in these curves, refer to the working plot. The fairing of that data may be at fault.

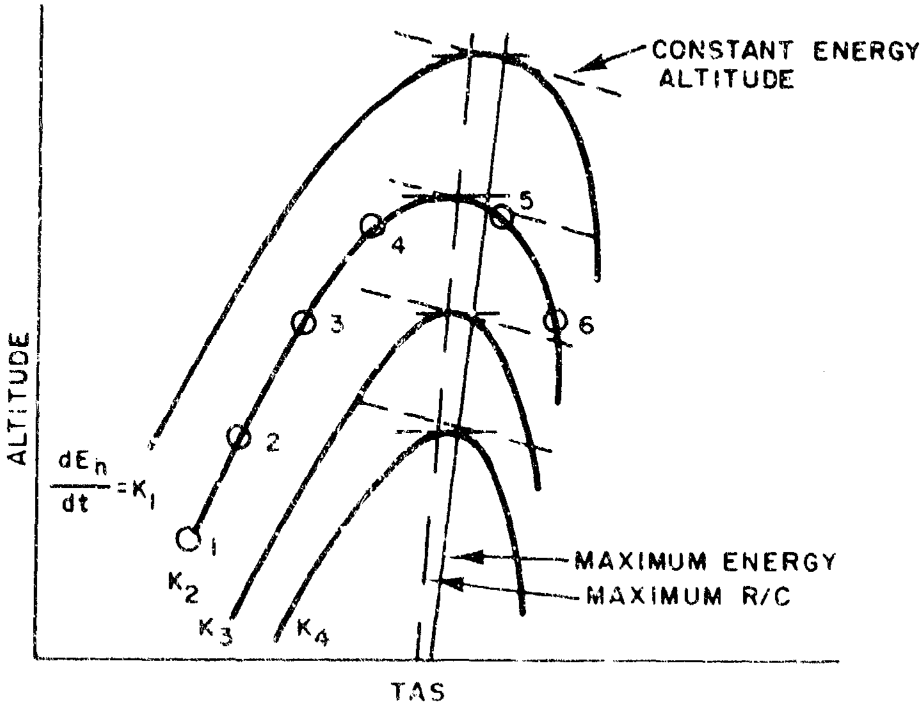


Figure 2
Specific Energy Analysis Plot

A climb schedule may be determined now. A line connecting the peak values of P_s is approximately a maximum rate of climb schedule. A line connecting the points at which the constant P_s lines are tangent with the energy height lines (i.e., points of peak energy height) is the maximum energy climb schedule. For the purpose of interpreting the climb schedule, it is convenient to scribe lines of constant Mach number on the Specific Energy Analysis plot.

Additional discussion of the maximum energy climb schedule is found on pages 156 through 165 of the Aircraft and Engine Performance manual.

SECTION XIII

COMPUTER DATA REDUCTION

COMPUTER DATA REDUCTION

ACCELERATION RUNS, CLIMB AND DESCENT PERFORMANCE

GENERAL

Analysis of level acceleration, climb, and descent data by digital computer routines is common practice at both military and contractor test facilities. The automatic data reduction process provides a rapid and accurate method of obtaining corrected performance results from raw recorded data.

DATA REDUCTION ROUTINE

The data reduction routines used by the NATC Computer Services Division (CSD) are mathematically similar to the methods described in applicable sections. The net thrust and induced drag corrections are added for level accelerations. The method requires inputs of observed airspeed, altitude, elapsed time, OAT, and fuel remaining or fuel used. Instrument corrections and position error corrections can be automatically applied from a correction subroutine. The engine net thrust correction is provided by a subroutine that is based on engine manufacturer's data.

DATA INPUT

Position error correction for each type airplane must be sent to CSD. Instrument corrections for each instrument used during the test must be submitted for each airplane for altitudes up to 50,000 ft. This information need only be submitted once for each airplane (if no instruments are changed between flights). See Figure 1 for sample format.

AIRPLANE TYPE ____________ BuNo ______________

(TYPE INSTRUMENT)/ (DATE OF CAL)		(TYPE INSTRUMENT)/ (DATE OF CAL)	
OBSERVED INST READING	INSTRUMENT CORRECTION	OBSERVED INST READING	INSTRUMENT CORRECTION

Figure 1
Sample Format for Instrument Calibration Submission

A performance data sheet for each acceleration run, climb, or descent must be submitted in the format shown in Figure 2. Data should be recorded every 4 sec (every other frame) for acceleration runs. For climbs it should be recorded every 4,000 ft (2,000 ft above 30,000 ft) or every 2 min, whichever comes first. For descents, record data every 5,000 ft. The elapsed time (Δt) is from start time (t_o) and should be measured in minutes and seconds.

Prior to submitting data to CSD for processing, a rough plot of V_o vs. time for acceleration runs or average rate of climb (or descent) vs. altitude for climbs (or descents) should be made to ensure you are submitting your data correctly. Any gross errors (such as failing to account for climb interruption) will be immediately obvious from these plots and unnecessary time delays will be avoided.

AIRPLANE TYPE ______________ BuNo ______________

PILOT ______________ DATE OF FLIGHT ______________

Standard Gross Weight ______________

Start Gross Weight (fuel used counter) ______________

Empty Gross Weight (fuel remaining system) ______________

Operational Weight for Climb Intercept ______________

Counts Per Gallon (6.8 lb/gal) ______________

t min + seconds	V_o (kt)	H_{p_o}	Fuel used or Fuel remaining (counts)	OAT_o (°C)	Fuel Flow (gpm or pph)

Figure 2
Sample Format for Computer Data Submission

DATA REDUCTION AND OUTPUT

After both data sheets (Figures 1 and 2) have been completed, these sheets will be delivered to CSD. Punched cards are then prepared by data reduction personnel; these cards are used for computer inputs. After the computer is programmed and the data reduction is completed, the results will be printed out automatically. The results will be presented in tabular form and will be plotted automatically. A typical automatic plot is shown in Figure 3.

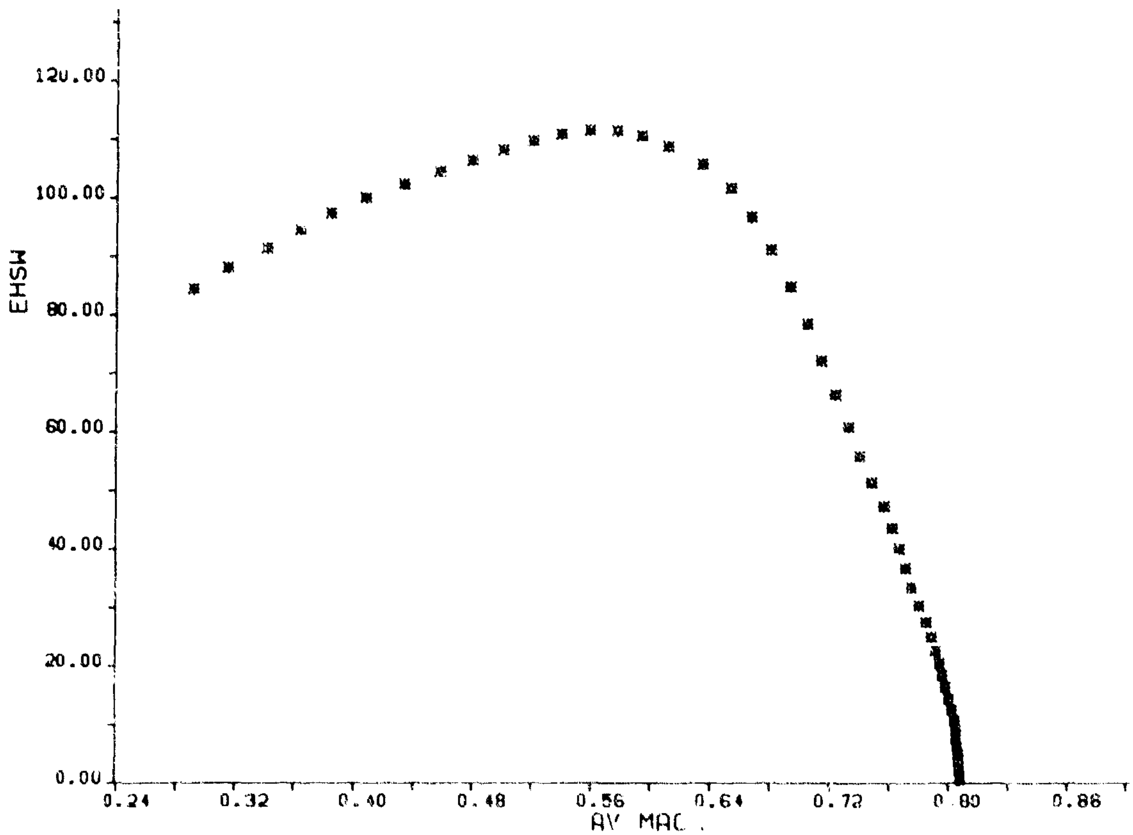


Figure 3
Typical Computer Printout from Accel Run

NOTATIONS

The notations used in the CSD data printout are:

HPO	-	Observed Altitude - Feet
HPI	-	Corrected for Instrument Error - Feet
HPC	-	Corrected for Position Error - Feet
PAI	-	Static Pressure from h_{p_i} - lbs/ft^2
PA	-	Static Pressure from h_{p_c} - lbs/ft^2
VO	-	Observed Airspeed - knots - V_o

VI	-	Corrected for Instrument Error - knots - V_i
VC	-	Corrected for Position Error - knots - V_c
QC	-	Impact pressure from V_c - lbs/ft^2
QCI	-	Impact pressure from V_i - lbs/ft^2
OAT	-	Outside Air Temperature - ^oC or ^oF
TI	-	Corrected for Instrument Error
MI	-	$f(q_{c_i}/p_{a_i})$
MACH	-	$f(q_c/p_a)$
TA	-	Temperature Ambient in ^oC
TAS	-	Temperature for Standard Day in ^oC
DT	-	$T_a - T_{a_s}$ ^oC
VTS	-	True Airspeed for Standard Day - V_{T_s}
VT	-	True Airspeed for Test Day - V_{T_t}
SRT (TA/TAS)	-	Square Root of T_a/T_{a_s}
FT	-	Observed Fuel Temperature
FTI	-	Corrected for Instrument Error
WF	-	Observed Fuel Flow
WFI	-	Corrected for Instrument Error
FUEL CN	-	Observed fuel used or fuel remaining (gals or lbs)
FCI	-	Corrected for Instrument Error
FUEL CN LB	-	Fuel counter converted to lbs
WT	-	TO Gross weight - Fuel used (lbs) = empty weight + fuel remaining
RPM	-	Observed RPM
RPMI	-	Corrected for Instrument Error

RPMC	-	Corrected for Tachometer Error
DFN	-	Thrust Correction - ΔF_n
DR/C 1	-	$(\Delta F_n \times V_{T_S} \times 101.3)/W_T$
R/CO	-	Observed rate of climb
(COSO)SQ	-	Cosine of the climb angle squared
C_L	-	Lift Coefficient
(B)SQE	-	Wing span^2 x Efficiency Factor (b^2e)
WS	-	Varying weight standard - calculated for fuel used or constant W_s
DR/C 2	-	$\dfrac{69877.0 \times V_{T_S} \times COS^2 \times (W_s^2 - W_T^2)}{b^2e \times \rho_a \times V_{T_S} \times W_s}$ - induced drag correction
RC/S	-	RCO + (DRC 1 + DRC 2) x .5
AV HPC	-	$(HPC_n + HPC_{n+1}) \times .5$
RCSW	-	RC/S x (WT/WS) - DR/C 2 - weight correction

SECTION XIV

TPS AIRPLANE DETAILS

AIRPLANE DETAILS

XIV - 2	T-2C
4	TA-4J
6	T-38A
8	A-7C
10	OV-1B

MODEL T-2C AIRPLANE

GENERAL

Loading Condition: Normal Trainer (one or two pilots)

T.O. Gross Weight: Approx 13,180 lb

Standard Gross Weight: 11,200 lb

Fuel Capacity: 696 gal/4,700 lb JP-5

Wing Span: 38.1 ft Area: 254.9 ft^2 Aspect Ratio: 5.07

POWER PLANT INFORMATION

Engine: Two J85-GE-4

Ratings:

Condition	Design Static SL Thrust (uninstalled-lb)	RPM %	Max EGT °C
MIL	2,950 (per engine)	101.7 maximum	732
NORM		99	718

CONFIGURATIONS

Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, thrust for level flight at trim airspeed.

Landing (L)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, Idle thrust.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 1/2, speedbrakes CLOSED, MIL thrust (takeoff trim).

MODEL TA-4J AIRPLANE

GENERAL

Internal Fuel Capacity: 660 gal/4,490 lb (JP-5)

T.O. Gross Weight: Approx 18,000 lb

Standard Gross Weight: 15,500 lb

POWER PLANT INFORMATION

Engine: Pratt and Whitney J-52-P6A/B

Condition	Design Static SL Thrust (uninstalled)-lb	RPM %	EGT °C
MIL	8,500	97-100	610
NORM	7,450	MIL-3	590

Engine: Pratt and Whitney J-52 - P-8B

Condition	Static Thrust (lb)	RPM %	EGT °C
MIL	9,300	97-100	650
NORM	8,250	MIL-3	595

CONFIGURATIONS

Cruise (CR)

Landing gear and flaps UP, speedbrakes (CLOSED), thrust for level flight at trim airspeed.

Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, thrust for level flight at trim airspeed.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 1/2, speedbrakes CLOSED, MIL thrust (takeoff trim).

MODEL T-38A AIRPLANE

GENERAL

Loading Condition: Normal Trainer (one or more pilots)

T.O. Gross Weight: Approx 12,000 lb

Standard Gross Weight: 9,950 lb

Fuel Capacity: 583 gal/3,750 lb (JP-4)

Wing Span: 25.25 ft Area (S): 170 ft^2 Aspect Ratio: 3.75

POWER PLANT INFORMATION

Engine: Two J85-GE-5

Ratings:

Condition	Design Static SL Thrust/Engine (uninstalled/ installed) lb	Maximum Steady State EGT °C	RPM %
MAX A/B	3,850/2,900	645	99.5-104
MIL	2,680/2,050	645	99.5-104

CONFIGURATIONS

Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Combat (CO)

Landing gear and flaps UP, speedbrakes CLOSED, MAX A/B thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, thrust for level flight at trim airspeed.

Landing (L)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, Idle thrust.

Wave-off (WO)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 60%, speedbrakes CLOSED, MAX A/B thrust.

MODEL A-7C AIRPLANE

GENERAL

Internal Fuel Capacity: 10,172 lb/1,496 gal (JP-5)

T.O. Gross Weight: Approx 29,500 lb

Standard Gross Weight: 26,000 lb

POWER PLANT INFORMATION

Engine: Pratt and Whitney TF-30-P408

Condition	Design Static SL Thrust (uninstalled)-lb	RPM %	EGT $^{\circ}C$
MIL	13,400	97.3	1160
NRM	12,[illegible]	-	10[illegible]

CONFIGURATIONS

Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, thrust for level flight (or 3 deg glide slope) at trim airspeed.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 25 deg, speedbrakes CLOSED, MIL thrust (takeoff trim).

MODEL OV-1B AIRPLANE

GENERAL

Loading Condition: Normal Observation

T.O. Gross Weight: Approx 13,300 lb

Standard Gross Weight: 12,500 lb

Fuel Capacity: (JP-4) 1,930 lb (no external)

3,880 lb (internal and external)

Wing Span: 48 ft

POWER PLANT INFORMATION

Engine: Two Lycoming T-53-L-7

Ratings:

Condition	SHP	RPM	Torque	EGT max	N_1 max
Takeoff	1,160	1,700	97 (max)	640°	101.5
MIL	1,160	1,700	97 (max)	640°	
NORM	900	1,600	80	620°	

Difference in TO and MIL is in throttle detent only. Five min max operation in TO, 30 min max operation in MIL.

CONFIGURATIONS

Cruise (CR)

Landing gear and flaps retracted, power for level flight.

Power (P)

Landing gear and flaps retracted, MIL power.

Glide (G)

Landing gear and flaps retracted, flight idle detent, min RPM.

Power Approach (PA)

Landing gear DOWN, flaps 45 deg, max RPM, torque as required for level flight or fixed as required.

Landing (L)

Landing gear DOWN, flaps 45 deg, max RPM, flight idle detent.

Wave-off (WO)

Landing gear DOWN, flaps 45 deg, TO power

Takeoff (TO)

Landing gear DOWN, flaps 15 deg, TO power, trim 1 deg, ND, 5 deg RWD, 5 deg NR.

SECTION XV

COCKPIT EVALUATION

REFERENCES

Section XV

1. Anthropometry of Naval Aviators, 1964, NAEC-ACEL-533.
2. Cockpit Anthropometric Survey of Model A-4C, A-6A, A-7E, AV-8A, F-4J, F-8D, and OV-10A Airplanes, ST-120R-71.
3. Investigation of A-4 Aircraft Escape System Clearance Envelope, ST-53R-72.
4. MIL-STD-1472A, Human Engineering Design Criteria for Military Systems, Equipment, and Facilities.
5. MIL-A-8806, General Specification for Acoustical Noise Levels in Aircraft.

COCKPIT EVALUATION

INTRODUCTION

The purpose of conducting a cockpit evaluation at TPS is to acquaint the student with the evaluation of human engineering design requirements as related to aircraft testing. Because the cockpit is the focal point of the man-machine interface, it is the area which should receive maximum human engineering design emphasis. Unfortunately, many individual cockpit items receive little or no emphasis concerning human accommodation or compatibility. It is often not until all cockpit items are assembled into a mockup or actual cockpit during design and testing evaluations that human engineering design deficiencies become identifiable. Not until the working relationships of controls, displays, lighting, and cockpit environment are analyzed can an intelligent evaluation be conducted. Regardless of airplane performance and flying quality potential, the man in the cockpit must accomplish the transfer function of changing airplane potential into reality. Important variables which control this transfer function are (1) capabilities and limitations of the aviator and (2) the cockpit design which is the link between action and reaction performed by the pilot as a result of information received and processed.

The primary airplane mission must be emphasized during human engineering evaluation. Additionally, "worst case" events must be considered such as degraded system operation and in-flight emergencies.

TEST PROCEDURES

The most important general principle to keep in mind when performing any human engineering evaluation is the concept of individual differences. We all tend

to evaluate items such as controls and displays from a very subjective viewpoint, i.e., our own capabilities, limitations, and experience. The fact that pilots are all different must be thoroughly understood when evaluating cockpits. Test pilots, in particular, are perhaps not always entirely objective when it comes to admitting that particular human engineering design deficiencies are a problem as far as they are concerned. The following differences are among the most important in cockpit evalaution.

ANTHROPOMETRY

Body sizes vary considerably. Reference 1 is a compilation of 96 body dimensions based on measurements of 1,549 Naval Aviators. The data are presented in inches and centimeters as well as percentiles.

Detail airplane specifications generally require that cockpits accommodate 5th through 95th percentile sized aviators for older airplane cockpits and crew stations (prior to 1970) and that 3rd through 98th percentile be accommodated in newer cockpits (since 1970). It is generally assumed that if one's body measurements, such as height and weight, are 50th percentile (average) that all his dimensions will be 50th percentile; this is not true. Uniformity in body dimensions is very rare; e.g., it is doubtful that if a person's sitting eye height is 70th percentile that his functional reach will also be 70th percentile. It is important to know one's own percentile ranks of body dimensions.

Physiological Training Units (altitude chambers) are equipped with anthropometric measuring devices where you can be measured and have your measurements translated into percentile ranks. The most important dimensions relative to aircrew station design are:

1. Total sitting height.

2. Sitting eye height.

3. Sitting shoulder height.

4. Bideltoid diameter (shoulder width).

5. Functional reach (grasp between thumb and forefinger).

6. Fingertip reach ("push-button" reach with extended forefinger).

7. Buttocks-to-knee length (sitting).

Only by knowing your own various percentile ranks can you make relative judgments as to the overall anthropometric accommodation of a particular cockpit; e.g., if you know your functional reach is 35th percentile and that you cannot reach a particular control when fully restrained, you therefore know that anyone with a functional reach less than 35th percentile, when fully restrained, also cannot reach the control.

Equipments exist which can objectively measure anthropometric parameters such as reach distances, angles of vision, and ejection seat egress clearances (References 2 and 3).

Egress clearances in ejection seat cockpits are often jeopardized when modifications such as cameras, control boxes, or other equipments, are added to canopy rails, glare shields, etc. Human engineering personnel are prepared to use particular equipments to attain quantitative data in cockpit anthropometry evaluations.

<u>NOTE:</u> It is critical that the <u>Design Eye Position (DEP)</u> be the source of measurements for anthropometric evaluations. The DEP is the point in space where the pilot's eyes should be to see all displays and have adequate exterior

vision. To further define the DEP, other preliminary definitions are in order and are presented below.

a. Seat Reference Point is a center line intersection of the seat back tangent line and seat surface.

b. Neutral Seat Reference Point (NSRP) is the location of the seat reference point when the seat is adjusted to the mid-point of vertical adjustment; e.g., with 5 in. of vertical seat travel available, the seat would be adjusted to 2.5 in. above the lower limit.

The DEP is then defined as the point in space located at the sitting eye height dimension of the 50th percentile average aviator (31.5 in.) measured vertically above the NSRP and 13 in. measured horizontally forward of the seat back tangent line. All anthropometric evaluations must originate at the DEP. Whatever the size of the individual evaluating items, such as control reach, display visibility, or cockpit space accommodation, the seat must be adjusted to place his eyes at the DEP. The necessity of adjusting the eyes to the DEP when making anthropometric evaluations is more critical now than ever with the increasing emphasis on heads-up displays and other optical devices which require strict adherence to line-of-sight criterion.

As with all human engineering evaluations, anthropometry must be checked against "worst case" conditions. An example of a "worst case" condition would be reaching for a critical control such as the emergency stores jettison when fully restrained (shoulder harness locked) and when under a high-g condition such as a catapult launch.

Additional items of anthropometric deficiency include insufficient sitting height, inability to reach rudder pedals or foot controls, inability to fit through emergency egress openings, etc.

CONTROL DESIGN

Controls must meet various criteria to be satisfactory; these criteria include:

1. Proper Location

The criticality of control function establishes the priority of location within a crew station. The most important controls should be the easiest to reach and manipulate. Controls should never be located such that the hand or arm manipulating the control is in the line of sight required to see the display effect or setting of the control.

2. Natural Direction-of-Motion Relationships

Actuating controls such as toggle switches forward or up should turn systems on. Turning rotary controls clockwise should increase system output. Standard direction-of-motion relationships should be adhered to in cockpit control actuation.

3. Shape Coding

Controls which may require manipulation without direct visual monitoring should feel different to the touch if they are near controls of dissimilar systems.

4. Inadvertent Actuation

Controls which can be activated incorrectly should be designed to prevent such activation either by electronic circuitry or mechanical guards; e.g., forward wing-sweep actuation during supersonic flight regimes which would potentially be damaging to airplane components should be electrically or mechanically prevented.

Care should be exercised in determining proper safeguards to prevent inadvertent actuation of controls, switches, etc., which might be actuated by flight clothing or items of personal equipment, such as survival vests, flotation devices, anti-exposure garments, etc.

5. Actuation Feedback

Controls should have proper tactile cues relative to actuation. One should "feel" the click of a toggle switch or push button without necessarily hearing it. Controls should have the proper resistance and range of displacement as specified by Reference 4.

DISPLAYS

The content and format of displays should be limited to essential information but should not require mental computation or translation to be usable. There must be provisions within displays to alert an operator to display failure if the failure is not immediately obvious.

The location and arrangement of displays shall be assigned priority relative to importance for normal and emergency operations. Other criteria, such as viewing distances, grouping, and transilluminated display requirements, are specified in Reference 4.

LABELING

Items of equipment which must be identified, manipulated, or located should be adequately labeled to permit efficient human performance. Blueprints which illustrate control panels often portray a straight-on-view. However, when the control panel is installed in a cockpit, it is often offset from direct line of sight. The three dimensional line-of-sight offset often results in labels (numbers, ON/OFF

legends, or other nomenclature) being obscured by the very controls to which they are related.

It is important to evaluate labeling legibility in low ambient light (dark) conditions as well as in daylight. If an item must be labeled for normal daylight use, it should be legible at night.

ENVIRONMENT

Heating, ventilation, and air conditioning shall be evaluated and compared with the criterion specified in Reference 4 or in the applicable specification listed in the detail specification of the airplane being evaluated. Hand-held instruments are available to measure temperature as well as relative humidity. Specifications generally require an Environmental Control System (ECS) to maintain between 60 and 80°F ambient temperature in a crew station and 10°F maximum differential between head and foot level.

Interior ambient air should also be sampled throughout the flight regime or mission profile of any aircraft. Carbon monoxide or other toxic fumes can be potential hazards, particularly during operations such as taxiing downwind, gun or rocket firing, and during refueling operations when directly behind a tanker.

Noise is the most serious and persistent problem among those associated with aircraft environment. The maximum allowable noise limits relative to aircraft type are described in Reference 5. It should be recognized that high noise levels of less intensity than those specified as physically damaging to hearing can produce human fatigue and degrade an aviator's effectiveness.

As a project officer, you should evaluate ambient exterior noise which maintenance or deck personnel are exposed to as a result of being in the immediate vicinity of the aircraft during ground operations. Maintenance personnel often neglect the required ear protection because hearing loss is a slow insidious process.

Various levels of instrumentation are available in evaluating the acoustical environment ranging from small pocket sized decibel meters to sophisticated type recording devices that record noise samples which can be analyzed in detail for various frequency bands.

NOTE: When conducting an interior noise survey, exercise any additional equipments which may increase the acoustic level, such as air conditioning or defogging systems, heater blowers, ambient air vents, and the extended configuration of in-flight refueling probes.

LIGHTING

A concerted effort in the evaluation of cockpit lighting usually identifies numerous lighting deficiencies. Often there is little emphasis on lighting evaluation. Typically, during night flights general lighting observations are made by crewmen who are busy flying or conducting other airborne tasks, thereby overlooking numerous lighting deficiencies.

A particular procedure which has been effective in static lighting evaluation is described below.

1. Get into the airplane attired in the complete compliment of proper flight clothing and equipment (take a tape recorder with you).

2. Have the canopy covered with an opaque cover preventing any ambient light from entering. This allows you to conduct the evaluation day or night.

3. Have electrical power supplied to the aircraft to enable interior light actuation.

4. Adjust your seat to place your eyes in the design eye position (or where you normally fly).

5. Allow your eyes to become adjusted to the dark (10 to 15 min).

6. Begin by locating the auxiliary light (if you can find it in the dark) and see if it is suitable for minimum illumination if all other lights were lost.

7. After the auxiliary light evaluation, systematically exercise all light controls in the cockpit. Vary the intensity, look for instrument lights on a particular rheostat which extinguish before others when adjusting from bright to OFF. Look for brightness imbalance such that, at a given light adjustment, some instruments may be too bright or too dim when most other instruments on that particular lighting control are at a reasonable intensity. Identify any glare or reflection which might possibly be shielded.

8. Verbally record on your tape recorder any deficiencies noted; this allows evaluation uninterrupted by turning on floodlights or flashlights to write down deficiencies which in turn would require readjustment of the eyes to low ambient light.

9. Adjust your seat to various positions to determine if lighting is sufficient throughout a typical range of particular eye locations. This may be one of the few times you evaluate lighting strictly for its own sake. Take as much time as is required to evaluate all lighting variations, legibility of labels, visibility of controls, etc.

SPECIFICATIONS

A partial listing of Human Engineering Specifications is listed below which may be of assistance in establishing Military Specification noncompliance.

MIL-STD-1472	Human Engineering Design Criteria for Systems, Equipment, and Facilities
MIL-STD-203E	Aircrew Station Controls and Displays for Fixed Wing Aircraft
MIL-STD-250C	Aircrew Station Controls and Displays for Rotary Wing Aircraft
MIL-STD-411D	Aircrew Station Signals
SD24J Vol. 1	General Specification for Design and Construction of Aircraft Weapons Systems, Fixed Wing Aircraft
SD24J Vol. 2	General Specification for Design and Construction of Aircraft Weapons Systems, Rotary Wing Aircraft

APPENDIXES

APPENDIXES

I. ICAO Standard Atmosphere Tables

II. Compressibility Correction

III. Airspeed - Altitude - Mach Number Crossplot

IV. Mach Number - Indicated Temperature - Recovery Factor - Ambient Temperature

V. Centigrade - Fahrenheit Temperature Conversion

VI. Calibrated Airspeed - Energy Relations

VII. Static Pressure versus Pressure Altitude*

VIII. Impact Pressure (q_c) versus Calibrated Airspeed*

*Derived from NASA TN D 822

STANDARD ATMOSPHERE TABLES

Sea Level Values

Pressure: 14.7 psi, 2,116 psf, 29.92 in Hg, 1,013 mb

Temperature: 59°F 15°C 518.69°Rankine 288.16°K

Density: .0023769 slugs/cu ft

Sonic Velocity: 661.48 kt, 1,116.45 fps

Gravitational Acceleration: 32.174 ft/sec/sec

Temperature lapse rate to 36,089 ft

-3.57°F per 1,000 ft -1.98°C per 1,000 ft

Sonic Velocity:

$$a = a_{sl} \left(\frac{T}{T_{sl}} \right)^{1/2}$$

Tropopause - at 36,089 ft geopotential altitude, T = 216.66°K

Density Ratio $\sigma = \frac{\rho}{\rho_{sl}}$

Temperature Ratio $\theta = \frac{T}{T_{sl}}$

Pressure Ratio $\delta = \frac{P}{P_{sl}}$

Viscosity Ratio $\phi = \frac{\mu}{\mu_{sl}}$

STANDARD ATMOSPHERE

H (FEET)	P_s (" Hg)	δ	$1/\delta$	T_s (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
-1000	31.018	1.0366	.9646	290.15	663.73	1.0069	1.0034	1.0295	1.0147	.9855	.9679	.9613	1.005	1.028
-900	30.907	1.0329	.9682	289.95	663.52	1.0062	1.0031	1.0266	1.0132	.9870	.9711	.9652		1.025
-800	30.796	1.0292	.9715	289.74	663.27	1.0055	1.0027	1.0236	1.0117	.9884	.9742	.9693	1.004	1.021
-700	30.686	1.0255	.9750	289.54	663.07	1.0048	1.0024	1.0206	1.0103	.9898	.9774	.9727		1.018
-600	30.575	1.0218	.9786	289.34	662.97	1.0041	1.0021	1.0176	1.0088	.9913	.9807	.9767	1.003	1.015
-500	30.465	1.0182	.9821	289.14	662.60	1.0034	1.0017	1.0147	1.0073	.9928	.9836	.9805		1.013
-400	30.356	1.0145	.9856	288.97	662.41	1.0028	1.0014	1.0117	1.0059	.9941	.9870	.9843	1.002	1.011
-300	30.247	1.0108	.9892	288.77	662.14	1.0021	1.0010	1.0088	1.0044	.9956	.9902	.9882		1.008
-200	30.138	1.0072	.9928	288.56	661.94	1.0014	1.0007	1.0058	1.0029	.9971	.9935	.9922	1.001	1.005
-100	30.029	1.0036	.9963	288.36	661.69	1.0007	1.0003	1.0029	1.0015	.9985	.9967	.9961		1.003
0	29.921	1.0000	1.0000	288.16	661.48	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.000	1.000
100	29.811	.9963	1.0036	287.96	661.28	.9993	.9997	.9970	.9985	1.0015	1.0033	1.0040		.9971
200	29.705	.9927	1.0072	287.76	661.02	.9986	.9993	.9941	.9971	1.0029	1.0066	1.0080	.999	.9943
300	29.598	.9892	1.0109	287.55	660.82	.9979	.9990	.9912	.9956	1.0044	1.0099	1.0121		.9918
400	29.491	.9856	1.0145	287.35	660.55	.9972	.9986	.9883	.9942	1.0058	1.0132	1.0161	.998	.9893
500	29.384	.9820	1.0182	287.18	660.36	.9966	.9983	.9854	.9927	1.0074	1.0165	1.0200		.9864
600	29.278	.9785	1.0219	286.98	660.09	.9959	.9979	.9825	.9912	1.0089	1.0198	1.0241	.997	.9834
700	29.172	.9749	1.0256	286.78	659.89	.9952	.9976	.9796	.9898	1.0103	1.0232	1.0282		.9807
800	29.066	.9714	1.0294	286.58	659.63	.9945	.9972	.9768	.9883	1.0118	1.0265	1.0323	.996	.9779
900	28.960	.9679	1.0331	286.37	659.43	.9938	.9969	.9739	.9869	1.0133	1.0300	0364		.9752
1000	28.855	.9643	1.0369	286.17	659.23	.9931	.9966	.9710	.9854	1.0148	1.0334	1.0406	.995	.9725
1100	28.750	.9608	1.0407	285.97	658.97	.9924	.9962	.9682	.9840	1.0163	1.0368	1.044		.9695
1200	28.646	.9573	1.0445	285.77	658.77	.9917	.9959	.9653	.9825	1.0178	1.0402	[illegible]	.994	.9671
1300	28.542	.9539	1.0483	285.60	658.50	.9911	.9955	.9625	.9811	1.0193	1.0436	[illegible]		.9644
1400	28.438	.9504	1.0521	285.39	658.31	.9904	.9952	.9596	.9796	1.0208	1.0471	1.0573	.993	.9617
1500	28.334	.9469	1.0559	285.19	658.04	.9897	.9948	.9568	.9782	1.0223	1.0505	1.0615		.9595
1600	28.231	.9435	1.0598	284.99	657.84	.9890	.9945	.9540	.9767	1.0239	1.0540	1.0658	.991	.9573
1700	28.128	.9400	1.0637	284.79	657.58	.9883	.9941	.9511	.9753	1.0255	1.0575	1.0700		.9546
1800	28.025	.9366	1.0676	284.59	657.39	.9876	.9938	.9483	.9739	1.0269	1.0610	1.0743	.990	.9519
1900	27.923	.9332	1.0715	284.39	657.12	.9869	.9934	.9455	.9724	1.0284	1.0645	1.0787		.9492
2000	27.821	.9298	1.0754	284.18	656.92	.9862	.9931	.9427	.9710	1.0299	1.0681	1.0831	.989	.9464
2100	27.719	.9264	1.0794	284.01	656.72	.9856	.9928	.9399	.9695	1.0315	1.0717	1.0874		.9435
2200	27.617	.9230	1.0834	283.81	656.45	.9849	.9924	.9371	.9681	1.0330	1.0752	1.0918	.988	.9413
2300	27.516	.9196	1.0873	283.61	656.26	.9842	.9921	.9344	.9666	1.0346	1.0788	1.0962		.9387
2400	27.415	.9162	1.0913	283.41	655.99	.9835	.9917	.9316	.9652	1.0361	1.0823	1.1006	.987	.9361
2500	27.315	.9129	1.0956	283.20	655.79	.9828	.9914	.9288	.9638	1.0376	1.0860	1.1050		.9334
2600	27.214	.9095	1.0994	283.00	655.53	.9821	.9910	.9261	.9623	1.0392	1.0895	1.1095	.986	.9308
2700	27.114	.9062	1.1034	282.80	655.33	.9814	.9907	.9233	.9609	1.0407	1.0932	1.1140		.9282
2800	27.015	.9028	1.1075	282.60	655.07	.9807	.9903	.9205	.9595	1.0422	1.0968	1.1184	.985	.9256
2900	26.915	.8995	1.1116	282.41	654.87	.9801	.9900	.9178	.9581	1.0437	1.1006	1.1230		.9230

H (FEET)	P_a (" Hg)	δ	$1/\delta$	T_a (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$\frac{1}{\delta}\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
3000	[illegible]	.8962	1.1157	282.22	654.59	.9734	.9896	.9151	.9566	1.0454	1.1042	1.1275	.984	.9203
3100	[illegible]	.8929	1.1199	282.02	654.35	.9787	.9893	.9123	.9552	1.0469	1.1079	1.1321		.9177
3200	26.619	.8896	1.1240	281.82	[illegible]	[illegible]	.9889	.9096	.9538	1.0484	1.1116	1.1367	.983	.9151
3300	26.521	.8863	1.1282	281.62	[illegible]	.9773	.9886	.9069	.9523	[illegible]	1.1153	1.1412		.9130
3400	[illegible]	.8830	1.1323	281.42	[illegible]	[illegible]	.9882	.9042	.9509	1.0516	1.1190	1.1458	.981	.9109
3500	[illegible]	.8798	1.1366	281.22	[illegible]	[illegible]	.9879	.9015	.9495	1.0532	1.1228	1.1506		.9083
3600	[illegible]	.8765	1.1407	281.01	[illegible]	[illegible]	.9875	.8988	[illegible]	[illegible]	1.1265	1.1552	.980	.9057
3700	[illegible]	[illegible]	1.1450	280.86	[illegible]	.9746	[illegible]	.8961	[illegible]	1.0564	1.1304	[illegible]		.9031
3800	[illegible]	[illegible]	1.1492	280.64	[illegible]	[illegible]	[illegible]	.8934	[illegible]	[illegible]	1.1342	[illegible]	.975	.9006
3900	[illegible]	.8668	1.1535	280.44	[illegible]	.9732	.9865	.8907	.9438	1.0595	1.1380	[illegible]		.8980
4000	25.841	.8636	1.1578	280.24	652.34	.9725	.9862	.8880	.9424	1.0611	1.1417	1.1743	.978	.8955
4100	25.746	.8604	1.1621	280.03	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	1.0627	1.1457	1.1789		.8930
4200	25.650	.8572	1.1665	279.83	[illegible]	[illegible]	.9855	[illegible]	.9395	1.0644	1.1496	1.1838	.977	.8905
4300	25.555	.8540	1.1705	279.63	[illegible]	.9704	.9851	.8801	.9381	1.0660	1.1534	1.1886		.8879
4400	25.460	.8509	1.1752	279.43	651.42	.9697	.9848	.8774	.9367	1.0676	1.1574	1.1936	.976	.8853
4500	25.365	.8477	1.1796	279.26	651.15	.9691	.9844	.8748	.9353	1.0692	1.1612	1.1983		.8828
4600	25.270	.8445	1.1840	279.05	[illegible]	.9684	.9841	.8721	.9339	1.0708	1.1652	1.2033	.975	.8805
4700	25.176	.8414	1.1884	278.85	[illegible]	.9677	.9837	.8695	.9325	1.0724	1.1691	1.2082		.8779
4800	25.082	.8382	1.1929	278.65	[illegible]	.9670	.9834	.8659	.9311	1.0740	1.1731	1.2132	.974	.8753
4900	24.989	.8351	1.1973	278.45	[illegible]	.9663	.9830	.8643	.9297	1.0756	1.1770	1.2180		.8732
5000	24.895	.8320	1.2018	278.25	[illegible]	.9655	.9827	.8616	.9283	1.0772	1.1811	1.2232	.972	.8711
5100	24.802	.8289	1.2063	278.05	[illegible]	.9649	.9823	.8590	.9269	1.0789	1.1850	1.2281		.8686
5200	[illegible]	.8258	1.2108	277.84	649.57	.9642	.9820	.8564	.9255	1.0805	1.1891	1.2333	.971	.8661
5300	[illegible]	.8227	1.2154	277.67	[illegible]	.9636	.9816	.8538	.9241	1.0821	1.1931	[illegible]		.8636
5400	24.525	.8196	1.2200	277.47	[illegible]	.9629	.9813	.8512	.9226	1.0839	1.1972	1.2434	.970	.8612
5500	24.433	.8166	1.2245	277.27	[illegible]	.9622	.9809	.8487	.9212	1.0855	1.2312	1.2485		.8587
5600	24.342	.8135	1.2292	277.07	648.64	.9615	.9805	.8461	.9198	1.0872	1.2054	1.2538	.969	.8562
5700	24.250	.8104	1.2338	276.86	648.39	.9608	.9802	.8435	.9184	1.0889	1.2094	1.2587		.8538
5800	24.159	.8074	1.2384	276.66	648.18	.9601	.9799	.8409	.9171	1.0904	1.2136	1.2640	.968	.8513
5900	24.068	.8044	1.2431	276.46	647.91	.9594	.9795	.8384	.9155	1.0922	1.2177	1.2692		.8488
6000	23.978	.8013	1.2478	276.26	647.71	.9587	.9792	.8358	.9143	1.0937	1.2219	1.2746	.967	.8464
6100	23.888	.7983	1.2525	276.09	647.45	.9581	.9788	.8333	.9129	1.0954	1.2260	1.2797		.8440
6200	23.798	.7953	1.2572	275.88	647.25	.9574	.9785	.8307	.9115	1.0971	1.2303	1.2851	.966	.8416
6300	23.708	.7923	1.2620	275.68	646.99	.9567	.9781	.8282	.9101	1.0988	1.2344	1.2904		.8392
6400	23.618	.7893	1.2668	275.48	646.79	.9560	.9778	.8257	.9087	1.1005	1.2387	1.2958	.965	.8366
6500	23.529	.7863	1.2716	275.28	[illegible]	.9553	.9774	.8231	.9073	1.1022	1.2429	1.3010		.8342
6600	23.440	.7834	1.2764	275.08	646.26	.9546	.9770	.8206	.9059	1.1039	1.2471	1.3064	.964	.8318
6700	23.352	.7804	1.2812	274.88	646.06	.9539	.9767	.8181	.9045	1.1056	1.2514	1.3119		.8295
6800	23.264	.7775	1.2861	274.67	645.80	.9532	.9763	.8156	.9031	1.1073	1.2557	1.3174	.963	.8269
6900	23.175	.7745	1.2910	274.50	645.60	.9526	.9760	.8131	.9017	1.1090	1.2601	1.3229		.8245

H (FEET)	P_H ("Hg)	δ	$\frac{1}{\delta}$	T_H (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$\frac{1}{\delta\sqrt{\theta}}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
7000	23.088	.7714	1.2954	274.30	645.34	.9519	.9756	.8106	.9004	1.1106	1.2643	1.3283	.962	.8223
7100	[illegible]	.7687	[illegible]	274.10	645.15	.9512	.9753	.8081	.8990	1.1123	1.2683	1.3340		.8202
7200	22.913	.7657	1.3058	273.90	644.95	.9505	.9749	.8055	.8975	1.1141	1.2731	1.3395	.960	.8182
7300	22.826	.7628	1.3108	273.69	644.66	.9498	.9746	.8032	.8962	1.1158	1.2775	1.3451		.8158
7400	22.739	.7599	1.3158	273.49	644.42	.9491	.9742	.8007	.8948	1.1176	1.2819	1.3506	.959	.8135
7500	22.653	.7570	1.3208	273.29	644.22	.9484	.9739	.7982	.8935	1.1192	1.2864	1.3561		.8111
7600	22.567	.7542	1.3258	273.09	[illegible]	.9477	.9735	.7957	.8921	1.1210	1.2908	1.3621	.958	.8087
7700	22.481	.7513	1.3309	272.92	643.76	.9471	.9732	.7933	.8907	1.1227	1.2953	1.3678		.8063
7800	22.395	.7484	1.3360	272.71	643.55	.9464	.9728	.7908	.8893	1.1245	1.2997	1.3734	.957	.8039
7900	22.310	.7456	1.3411	272.51	643.33	.9457	.9725	.7884	.8880	1.1261	1.3043	1.3793		.8016
8000	22.225	.7427	1.3462	272.31	643.08	.9450	.9721	.7859	.8865	1.1279	1.3087	1.3850	.956	.7992
8100	22.140	.7399	1.3514	272.11	642.85	.9443	.9718	.7835	.8852	1.1297	1.3133	1.3909		.7968
8200	22.055	.7371	1.3566	271.91	642.57	.9436	.9714	.7811	.8838	1.1315	1.3178	1.3966	.955	.7944
8300	21.971	.7343	1.3618	271.71	642.34	.9429	.9710	.7787	.8825	1.1331	1.3223	1.402		.7921
8400	21.887	.7314	1.3670	271.50	642.10	.9422	.9707	.7763	.8811	1.1349	1.3270	1.408	.954	.7898
8500	21.803	.7286	1.3723	271.33	641.84	.9416	.9703	.7739	.8797	1.1368	1.3316	1.414		.7879
8600	21.719	.7259	1.3776	271.13	641.64	.9409	.9700	.7715	.8784	1.1384	1.3361	1.420	.952	.7860
8700	21.636	.7231	1.3829	270.93	641.42	.9402	.9696	.7691	.8770	1.1403	1.3409	1.426		.7837
8800	21.553	.7203	1.3882	270.73	641.18	.9395	.9693	.7667	.8756	1.1421	1.3456	1.432	.951	.7813
8900	21.470	.7175	1.3935	270.52	640.91	.9382	.9689	.7643	.8743	1.1438	1.3503	1.438		.7789
9000	21.388	.7148	1.3990	270.32	640.72	.9381	.9685	.7619	.8729	1.1456	1.3550	1.444	.950	.7767
9100	21.305	.7120	1.4043	270.12	640.45	.9374	.9682	.7595	.8715	1.1474	1.3597	1.450		.7744
9200	21.223	.7093	1.4098	269.92	640.25	.9367	.9679	.7572	.8702	1.1492	1.3645	1.456	.949	.7722
9300	21.142	.7065	1.4152	269.75	639.99	.9361	.9675	.7548	.8688	1.1510	1.3693	1.462		.7699
9400	21.060	.7038	1.4207	269.54	639.72	.9354	.9671	.7525	.8675	1.1527	1.3740	1.469	.948	.7676
9500	20.979	.7011	1.4262	269.34	639.52	.9347	.9668	.7501	.8661	1.1546	1.3789	1.475		.7653
9600	20.899	.6984	1.4317	269.14	639.26	.9340	.9664	.7478	.8648	1.1563	1.3836	1.481	.947	.7633
9700	20.817	.6957	1.4372	268.94	639.00	.9333	.9661	.7454	.8634	1.1582	1.3886	1.487		.7609
9800	20.737	.6930	1.4428	268.74	638.79	.9326	.9657	.7431	.8621	1.1600	1.3934	1.494	.946	.7585
9900	20.658	.6903	1.4484	268.54	638.54	.9319	.9654	.7408	.8607	1.1618	1.3984	1.500		.7562
10000	20.577	.6877	1.4541	268.33	638.33	.9312	.9650	.7384	.8593	1.1637	1.4032	1.506	.945	.7539
10100	20.497	.6850	1.4597	268.13	638.13	.9305	.9647	.7361	.8580	1.1655	1.4082	1.513		.7516
10200	20.417	.6823	1.4654	267.93	637.90	.9298	.9643	.7338	.8566	1.1674	1.4131	1.519	.944	.7493
10300	20.338	.6797	1.4711	267.73	637.66	.9291	.9639	.7315	.8553	1.1692	1.4180	1.526		.7470
10400	20.259	.6771	1.4768	267.53	637.42	.9285	.9636	.7292	.8540	1.1710	1.4231	1.532	.942	.7447
10500	20.180	.6744	1.4826	267.33	637.13	.9278	.9632	.7269	.8526	1.1729	1.4281	1.539		.7425
10600	20.102	.6718	1.4884	267.13	636.90	.9271	.9629	.7246	.8513	1.1747	1.4332	1.545	.941	.7402
10700	20.024	.6692	1.4942	266.93	636.67	.9264	.9625	.7223	.8499	1.1766	1.4382	1.552		.7380
10800	19.946	.6666	1.5001	266.73	636.47	.9257	.9622	.7200	.8486	1.1784	1.4434	1.559	.940	.7359
10900	19.869	.6640	1.5059	266.53	636.21	.9251	.9616	.7178	.8472	1.1804	1.4485	1.566		.7337

H (FEET)	P_0 ("Hg)	δ	$1/\delta$	T_0 (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
11000	19.790	.6614	1.5118	266.38	635.94	.9244	.9614	.7155	.8459	1.1822	1.4533	1.572	.939	.7324
11100	19.713	.6588	1.5177	266.17	[illegible]	.9237	.9611	.7132	.8446	1.1840	1.4588	1.579		.7302
11200	19.636	.6562	1.5237	265.97	635.48	.9230	.9607	.7110	.8432	1.1860	1.4639	1.586	.938	.7280
11300	19.559	.6537	1.5297	265.77	635.28	.9223	.9604	.7087	.8419	1.1878	1.4692	1.593		.7258
11400	19.483	.6511	1.5357	265.57	635.03	.9216	.9600	.7065	.8406	1.1896	1.4743	1.599	.937	.7237
11500	19.407	.6486	1.5417	265.37	634.82	.9209	.9597	.7043	.8392	1.1916	1.4796	1.6067		.7215
11600	19.331	.6460	1.5478	265.16	634.56	.9202	.9593	.7020	.8379	1.1935	1.4848	1.6136	.936	.7193
11700	19.255	.6435	1.5539	264.99	634.29	.9196	.9589	.6998	.8366	1.1953	1.4901	1.6206		.7175
11800	19.179	.6410	1.5600	264.79	634.09	.9189	.9586	.6976	.8352	1.1973	1.4955	1.6276	.934	.7157
11900	19.104	.6384	1.5661	264.59	633.83	.9182	.9582	.6953	.8339	1.1992	1.5007	1.6345		.7136
12000	19.029	.6359	1.5723	264.39	633.63	.9175	.9579	.6931	.8326	1.2011	1.5062	1.6417	.933	.7114
12100	18.954	.6334	1.5785	264.19	633.36	.9168	.9575	.6909	.8312	1.2031	1.5115	1.6488		.7093
12200	18.879	.6309	1.5848	263.98	633.10	.9161	.9571	.6387	.8299	1.2050	1.5163	1.6557	.932	.7072
12300	18.805	.6285	1.5910	263.78	632.90	.9154	.9566	.6865	.8286	1.2069	1.5224	1.6631		.7050
12400	18.731	.6260	1.5973	263.58	632.64	.9147	.9564	.6843	.8273	1.2088	1.5277	1.6702	.931	.7028
12500	18.657	.6235	1.6036	263.41	632.44	.9141	.9561	.6821	.8259	1.2108	1.5333	1.6775		.7007
12600	18.583	.6210	1.6100	263.21	632.17	.9134	.9557	.6800	.8246	1.2127	1.5387	1.6846	.930	.6986
12700	18.510	.6186	1.6164	263.00	631.91	.9127	.9553	.6778	.8233	1.2146	1.5442	1.6920		.6965
12800	18.437	.6161	1.6228	262.80	631.71	.9120	.9550	.6756	.8220	1.2165	1.5498	1.6994	.929	.6943
12900	18.364	.6137	1.6293	262.60	631.45	.9113	.9546	.6735	.8207	1.2185	1.5553	1.7063		.6921
13000	18.291	.6113	1.6357	262.40	631.25	.9106	.9543	.6713	.8194	1.2204	1.5610	1.7142	.928	[illegible]
13100	18.219	.6089	1.6422	262.20	630.98	.9099	.9539	.6691	.8180	1.2225	1.5666	1.7217		.6876
13200	18.147	.6064	1.6488	262.00	630.72	.9092	.9535	.6670	.8167	1.2244	1.5722	1.7292	.927	.6858
13300	18.075	.6040	1.6554	261.82	630.52	.9086	.9532	.6648	.8154	1.2264	1.5779	1.7368		.6842
13400	18.003	.6016	1.6619	261.62	630.26	.9079	.9528	.6627	.8141	1.2284	1.5835	1.7442	.925	.6823
13500	17.931	.5992	1.6686	261.40	630.06	.9072	.9525	.6606	.8128	1.2303	1.5894	1.7521		.6804
13600	17.860	.5969	1.6752	261.22	629.79	.9065	.9521	.6584	.8115	1.2323	1.5950	1.7596	.924	.6783
13700	17.789	.5945	1.6820	261.02	629.53	.9058	.9517	.6563	.8102	1.2343	1.6003	1.7674		.6763
13800	17.718	.5921	1.6887	260.81	629.33	.9051	.9514	.6542	.8089	1.2362	1.6066	1.7750	.923	.6743
13900	17.647	.5898	1.6954	260.61	629.07	.9044	.9510	.6521	.8075	1.2382	1.6124	1.7828		.6722
14000	17.577	.5874	1.7022	260.41	628.87	.9037	.9507	.6500	.8062	1.2404	1.6184	1.7908	.922	.6702
14100	17.507	.5851	1.7090	260.24	628.60	.9031	.9503	.6479	.8049	1.2424	1.6241	1.7985		[illegible]
14200	17.437	.5827	1.7159	260.04	628.34	.9024	.9499	.6458	.8036	1.2444	1.6300	1.8064	.921	[illegible]
14300	17.367	.5804	1.7229	259.83	628.14	.9017	.9496	.6437	.8023	1.2464	1.6360	1.8144		[illegible]
14400	17.298	.5781	1.7297	259.63	627.88	.9010	.9492	.6416	.8010	1.2484	1.6419	1.8224	.920	[illegible]
14500	17.228	.5758	1.7367	259.43	627.61	.9003	.9488	.6395	.7997	1.2505	1.6478	1.8304		[illegible]
14600	17.159	.5735	1.7436	259.23	627.41	.8996	.9485	.6375	.7984	1.2525	1.6539	1.8386	.919	[illegible]
14700	17.090	.5712	1.7507	259.03	627.15	.8989	.9481	.6354	.7971	1.2545	1.6598	1.8464		[illegible]
14800	17.022	.5689	1.7577	258.83	626.95	.8982	.9478	.6333	.7958	1.2566	1.6660	1.8544	.917	[illegible]
14900	16.953	.5666	1.7648	258.65	626.69	.8976	.9474	.6313	.7945	1.2587	1.6720	1.8629		[illegible]

H (FEET)	P (in Hg)	δ	1/δ	T (°K)	a (KNOTS)	θ	√θ	σ	√σ	1/√σ	√θ/δ	(1/δ)√θ	φ	δ/(φ√θ)
15000	16.875	.5645	1.7714	258.45	626.42	.8969	.9470	.6292	.7932	1.2607	1.6781	1.8712	.916	.6507
15100	16.817	.5620	1.7791	258.25	626.22	.8962	.9467	.6271	.7920	1.2626	1.6843	1.8784		.6487
15200	16.75[illegible]	.5598	1.7863	258.05	625.96	.8955	.9463	.6251	.7907	1.2647	1.6904	1.8857	.915	.6467
15300	16.692	.5575	1.7935	257.85	625.69	.8948	.9459	.6231	.7894	1.2668	1.6965	1.8949		.6447
15400	16.635	.5553	1.8008	257.64	625.50	.8941	.9456	.6210	.7881	1.2689	1.7029	1.9047	.914	.6427
15500	16.548	.5530	1.8081	257.44	625.23	.8934	.9452	.6190	.7868	1.2710	1.7090	1.9129		.6407
15600	16.491	.5508	1.8154	257.24	625.01	.8927	.9449	.6170	.7855	1.2731	1.7154	1.9216	.913	.6386
15700	16.4[illegible]	.5485	1.8228	257.03	624.77	.8921	.9445	.6149	.7842	1.2752	1.7217	1.9301		.6367
15800	16.348	.5463	1.8302	256.87	624.50	.8914	.9441	.6128	.7829	1.2773	1.7279	1.9386	.912	.6347
15900	16.2[illegible]	.5441	1.8376	256.66	624.3[illegible]	.8907	.9438	.6108	.7816	1.2794	1.7344	1.9473		.6330
16000	16.216	.5419	1.8451	256.46	624.16	.8900	.9434	.6089	.7804	1.2814	1.7407	1.9559	.910	.6314
16100	16.150	.5397	1.8526	256.26	623.76	.8893	.9430	.6069	.7791	1.2835	1.7470	1.9645		.6294
16200	16.085	.5375	1.8601	256.05	[illegible]	.8886	.9427	.6049	.7778	1.2857	1.7536	1.9735	.909	.6274
16300	16.019	.5356	1.8677	255.86	[illegible]	.8879	.9423	.6029	.7765	1.2878	1.7600	1.9822		.6254
16400	15.954	.5332	1.8753	255.66	[illegible]	.8872	.9419	.6009	.7752	1.2900	1.7664	1.9909	.908	.6234
16500	15.8[illegible]	.5310	1.8830	255.48	622.5[illegible]	.8866	.9416	.5989	.7740	1.2920	1.7731	2.0001		.6214
16600	15.825	.5288	1.8907	255.28	622.98	.8859	.9412	.5970	.7727	1.2942	1.7795	2.0090	.907	.6195
16700	15.760	.5267	1.8984	255.08	622.5[illegible]	.8852	.9408	.5950	.7714	1.2963	1.7861	2.0178		.6175
16800	15.696	.5245	1.9062	254.88	622.1[illegible]	.8845	.9405	.5931	.7701	1.2985	1.7928	2.0270	.906	.6156
16900	15.632	.5224	1.9140	254.68	621.85	.8838	.9401	.5911	.7689	1.3005	1.7995	2.0361		.6136
17000	15.568	.5203	1.9218	254.47	621.58	.8831	.9397	.5891	.7676	1.3028	1.8060	2.0451	.905	.6117
17100	15.505	.5182	1.9297	254.27	621.32	.8824	.9394	.5872	.7663	1.3050	1.8128	2.0541		.6098
17200	15.441	.5160	1.9376	254.07	621.12	.8817	.9390	.5853	.7651	1.3070	1.8195	2.0636	.903	.6080
17300	15.378	.5139	1.9456	253.90	620.90	.8811	.9386	.5833	.7638	1.3092	1.8263	2.0728		.6067
17400	15.315	.5118	1.9536	253.70	620.[illegible]	.8804	.9383	.5814	.7625	1.3115	1.8331	2.0823	.902	.6047
17500	15.252	.5097	1.9617	253.49	620.3[illegible]	.8797	.9379	.5794	.7612	1.3137	1.8398	2.0916		.6028
17600	15.190	.5076	1.9697	253.29	620.13	.8790	.9375	.5775	.7600	1.3153	1.8467	2.1010	.901	.6008
17700	15.127	.5054	1.9778	253.09	619.93	.8783	.9372	.5756	.7587	1.3180	1.8537	2.1106		.5989
17800	15.065	.5035	1.9860	252.89	619.67	.8776	.9368	.5737	.7575	1.3201	1.8605	2.1201	.900	.5970
17900	15.003	.5014	1.9942	252.69	619.40	.8769	.9364	.5718	.7562	1.3224	1.8674	2.1295		.5952
18000	14.942	.4993	2.0024	252.49	619.21	.8762	.9361	.5699	.7549	1.3247	1.8745	2.1393	.899	.5933
18100	14.880	.4973	2.0107	252.31	618.95	.8756	.9357	.5680	.7537	1.3268	1.8815	2.1490		.5916
18200	14.819	.4952	2.0191	252.11	618.68	.8749	.9353	.5661	.7524	1.3291	1.8885	2.1587	.898	.5896
18300	14.758	.4932	2.0274	251.91	618.4[illegible]	.8742	.9350	.5642	.7511	1.3314	1.8957	2.1684		.5880
18400	14.697	.4912	2.0358	251.71	618.22	.8735	.9346	.5623	.7499	1.3335	1.9027	2.1781	.896	.5863
18500	14.636	.4891	2.0442	251.51	617.96	.8728	.9342	.5604	.7486	1.3358	1.9098	2.1882		.5846
18600	14.576	.4871	2.0527	251.30	617.76	.8721	.9339	.5585	.7474	1.3380	1.9171	2.1983	.895	.5827
18700	14.515	.4851	2.0613	251.10	617.49	.8714	.9335	.5567	.7461	1.3403	1.9242	2.2081		.5809
18800	14.455	.4831	2.0698	250.90	617.[illegible]	.8707	.9331	.5548	.7449	1.3425	1.9314	2.2182	.894	.5791
18900	14.395	.4811	2.0784	250.73	617.05	.8701	.9328	.5529	.7436	1.3448	1.9388	2.2284		.5772

h (FEET)	P_h ("H_g)	δ	$1/\delta$	Tc (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
19000	14.336	.4791	2.0871	250.53	[illegible]	.8694	.9324	.5511	.7424	1.3470	1.9460	2.23[illegible]	.893	.5754
19100	14.276	.4771	2.0958	250.32	[illegible]	.8687	.9320	.5492	.7411	1.3493	1.9533	2.24[illegible]		.5735
19200	14.217	.4751	2.1045	250.12	[illegible]	.8680	.9317	.5474	.7399	1.3515	1.9606	2.25[illegible]	.892	.5716
19300	14.158	.4731	2.1133	249.92	516.[illegible]	.8673	.9313	.5455	.7386	1.3539	1.9682	2.269		.5698
19400	14.099	.4712	2.1221	249.72	515.77	.8666	.9309	.5437	.7374	1.3561	1.9756	2.279	.891	.5680
19500	14.040	.4692	2.1310	249.52	[illegible]	.8659	.9306	.5419	.7361	1.3585	1.9832	2.290		.5665
19600	13.982	.4673	2.1399	249.32	[illegible]	.8652	.9302	.5400	.7349	1.3607	1.9906	2.300	.889	.5650
19700	13.923	.4653	2.1489	249.14	[illegible]	.8646	.9298	.5382	.7337	1.3630	1.9981	2.311		.5631
19800	13.865	.4634	2.1579	248.94	[illegible]	.8639	.9294	.5364	.7324	1.3654	2.0056	2.321	.888	.5612
19900	13.807	.4614	2.1669	248.74	[illegible]	.8632	.9291	.5346	.7312	1.3676	2.0133	2.332		.5394
20000	13.750	.4595	2.1760	248.54	[illegible] 32	.8625	.9287	.5328	.7299	1.3701	2.0207	2.343	.887	.5576
20100	13.692	.4576	2.1852	248.34	[illegible]	.8618	.9283	.5310	.7287	1.3723	2.0285	2.353		.5559
20200	13.635	.4557	2.1944	248.1[illegible]	[illegible]	.8611	.9280	.5292	.7275	1.3746	2.0364	2.364	.886	.5541
20300	13.575	.4537	2.2036	247.[illegible]	[illegible]	.8604	.9276	.5274	.7262	1.3770	2.0441	2.375		.5524
20400	13.5[illegible]	.4518	2.2129	247.[illegible]	[illegible]	.8597	.9272	.5256	.7250	1.3793	2.0518	2.386	.885	.5506
20500	13.464	.4500	2.2222	247.56	[illegible]	.8591	.9269	.5238	.7238	1.3816	2.0598	2.397		.5488
20600	13.407	.4481	2.2315	247.3[illegible]	[illegible]	.8584	.9265	.5221	.7225	1.3841	2.0676	2.408	.884	.5470
20700	13.351	.4462	2.2410	247.1[illegible]	[illegible]	.8577	.9261	.5202	.7213	1.3864	2.0754	2.419		.5452
20800	13.295	.4443	2.2504	246.95	[illegible]	.8570	.9257	.5185	.7201	1.3887	2.0833	2.431	.883	.5434
20900	13.239	.4424	2.2600	246.75	512.14	.8563	.9254	.5167	.7188	1.3912	2.0914	2.442		.5420
21000	13.183	.4405	2.2695	246.55	511.0[illegible]	.8556	.9250	.5149	.7176	1.3935	2.0994	2.453	.881	.5406
21100	13.128	.4387	2.2792	246.35	511.[illegible]	.8549	.9246	.5132	.7164	1.3959	2.1074	2.465		.5388
21200	13.072	.4369	2.2888	246.15	[illegible]	.8542	.9243	.5114	.7152	1.3982	2.1156	2.475	.880	.5371
21300	13.017	.4350	2.2985	245.97	511.15	.8536	.9239	.5097	.7139	1.4006	2.1236	2.4[illegible]		.5354
21400	12.962	.4332	2.3082	245.77	510.[illegible]	.8529	.9235	.5079	.7127	1.4031	2.1317	2.499	.879	.5336
21500	12.907	.4313	2.3180	245.57	[illegible]	.8522	.9231	.5062	.7115	1.4055	2.1398	2.511		.5320
21600	12.852	.4295	2.3279	245.37	[illegible]	.8515	.9228	.5044	.7103	1.4079	2.1482	2.522	.878	.5303
21700	12.798	.4277	2.3378	245.[illegible]	[illegible]	.8508	.9224	.5027	.7090	1.4104	2.1565	2.534		.5286
21800	12.744	.4259	2.3478	244.96	[illegible]	.8501	.9220	.5010	.7078	1.4123	2.1647	2.546	.877	.5268
21900	12.690	.4241	2.3578	244.76	[illegible]	.8494	.9216	.4993	.7066	1.4152	2.1730	2.558		.5254
22000	12.636	.4223	2.3678	244.56	[illegible]	.8487	.9213	.4975	.7054	1.4176	2.1815	2.571	.875	.5239
22100	12.582	.4205	2.378	244.36	509.1[illegible]	.8481	.9209	.4952	.7042	1.4201	2.1899	2.582		.5222
22200	12.529	.4187	2.388	244.1[illegible]	[illegible]	.8474	.9205	.4941	.7030	1.4225	2.1983	2.594	.874	.5205
22300	12.475	.4169	2.398	243.[illegible]	[illegible]	.8467	.9202	.4923	.7017	1.4251	2.2070	2.606		.5188
22400	12.422	.4151	2.408	243.78	[illegible]	.8450	.9190	.4906	.7005	1.4275	2.2155	2.618	.873	.5171
22500	12.369	.4134	2.418	243.58	[illegible]	.8453	.9194	.4890	.6993	1.4300	2.2240	2.631		.5154
22600	12.316	.4116	2.429	243.38	[illegible]	.8446	.9190	.4873	.6981	1.4325	2.2325	2.643	.872	.5136
22700	12.264	.4098	2.439	243.18	[illegible]	.8439	.9187	.4856	.6969	1.4349	2.2414	2.656		.5119
22800	12.211	.4081	2.450	242.98	[illegible] 43	.8432	.9183	.4340	.6957	1.4374	2.2500	2.663	.871	.5102
22900	12.159	.4063	2.460	242.80	507.17	.8426	.9179	.4323	.6945	1.4399	2.2587	2.680		.5085

H (FEET)	P_o ("H_g)	δ	$1/\delta$	T_o (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
23000	12.107	.4045	2.471	242.60	606.90	.8419	.9175	.4806	.6933	1.4424	2.2674	2.693	.870	.5066
23100	12.055	.4025	2.481	242.40	606.70	.8412	.9172	.4789	.6921	1.4449	2.2764	2.706		.5052
23200	12.003	.4011	2.492	242.20	606.44	.8405	.9168	.4773	.6909	1.4474	2.2853	2.719	.869	.5037
23300	11.952	.3994	2.503	242.00	606.18	.8398	.9164	.4756	.6897	1.4499	2.2941	2.731		.5022
23400	11.901	.3977	2.514	241.80	605.91	.8391	.9160	.4740	.6885	1.4524	2.3030	2.744	.867	.5005
23500	11.849	.3960	2.525	241.59	605.71	.8384	.9157	.4723	.6873	1.4550	2.3122	2.757		.4986
23600	11.798	.3943	2.535	241.39	605.45	.8377	.9153	.4707	.6861	1.4575	2.3212	2.770	.865	.4969
23700	11.748	.3925	2.546	241.22	605.18	.8371	.9149	.4690	.6849	1.4601	2.3302	2.783		.4957
23800	11.697	.3909	2.557	241.02	604.92	.8364	.9145	.4674	.6837	1.4626	2.3392	2.797	.864	.4945
23900	11.646	.3892	2.569	240.82	604.72	.8357	.9142	.4657	.6825	1.4652	2.3486	2.810		.4934
24000	11.595	.3875	2.580	240.61	604.46	.8350	.9138	.4641	.6813	1.4678	2.3578	2.823	.863	.4913
24100	11.546	.3859	2.591	240.41	604.19	.8343	.9134	.4625	.6801	1.4704	2.3669	2.837		.4896
24200	11.495	.3842	2.602	240.21	603.93	.8335	.9130	.4609	.6789	1.4730	2.3762	2.850	.862	.4880
24300	11.446	.3825	2.613	240.01	603.66	.8329	.9126	.4593	.6777	1.4756	2.3854	2.864		.4864
24400	11.397	.3809	2.625	239.81	603.46	.8322	.9123	.4577	.6765	1.4782	2.3951	2.877	.861	.4848
24500	11.347	.3792	2.636	239.63	603.20	.8315	.9119	.4560	.6753	1.4808	2.404	2.891		.4832
24600	11.298	.3776	2.648	239.43	602.93	.8309	.9115	.4544	.6742	1.4832	2.413	2.905	.860	.4816
24700	11.249	.3759	2.659	239.23	602.67	.8302	.9111	.4528	.6730	1.4859	2.423	2.919		.4799
24800	11.200	.3742	2.671	239.03	602.47	.8295	.9108	.4512	.6718	1.4885	2.433	2.933	.859	.4783
24900	11.152	.3727	2.683	238.83	602.21	.8288	.9104	.4497	.6706	1.4912	2.442	2.947		.4767
25000	11.103	.3710	2.694	238.63	601.94	.8281	.9100	.4481	.6694	1.4939	2.452	2.961	.858	.4752
25100	11.055	.3694	2.706	238.43	601.68	.8274	.9096	.4465	.6682	1.4966	2.461	2.975		.4735
25200	11.006	.3678	2.719	238.22	601.48	.8267	.9093	.4449	.6671	1.4990	2.471	2.990	.857	.4721
25300	10.958	.3662	2.730	238.03	601.22	.8261	.9089	.4433	.6659	1.5017	2.481	3.004		.4709
25400	10.911	.3646	2.742	237.85	600.95	.8254	.9085	.4418	.6647	1.5044	2.491	3.018	.855	.4696
25500	10.863	.3630	2.754	237.65	600.69	.8247	.9081	.4402	.6635	1.5072	2.501	3.033		.4680
25600	10.815	.3614	2.766	237.44	600.42	.8240	.9077	.4387	.6623	1.5099	2.511	3.047	.854	.4663
25700	10.768	.3598	2.778	237.24	600.22	.8233	.9074	.4371	.6612	1.5124	2.521	3.062		.4647
25800	10.721	.3583	2.790	237.04	599.96	.8226	.9070	.4355	.6600	1.5152	2.531	3.077	.853	.4632
25900	10.674	.3567	2.803	236.84	599.69	.8219	.9066	.4340	.6588	1.5179	2.541	3.092		.4616
26000	10.627	.3551	2.815	236.64	599.43	.8212	.9062	.4324	.6576	1.5207	2.551	3.106	.852	.4601
26100	10.580	.3536	2.827	236.45	599.17	.8206	.9058	.4309	.6565	1.5232	2.561	3.121		.4588
26200	10.534	.3520	2.840	236.26	598.97	.8199	.9055	.4294	.6553	1.5260	2.572	3.137	.850	.4575
26300	10.487	.3505	2.852	236.06	598.70	.8192	.9051	.4278	.6541	1.5288	2.582	3.152		.4559
26400	10.441	.3489	2.865	235.86	598.44	.8185	.9047	.4263	.6530	1.5314	2.592	3.167	.849	.4543
26500	10.395	.3474	2.878	235.66	598.17	.8178	.9043	.4248	.6518	1.5342	2.602	3.182		.4527
26600	10.349	.3459	2.891	235.46	597.91	.8171	.9039	.4233	.6506	1.5370	2.613	3.198	.848	.4512
26700	10.304	.3443	2.903	235.25	597.71	.8164	.9036	.4218	.6495	1.5396	2.623	3.214		.4496
26800	10.258	.3428	2.916	235.05	597.45	.8157	.9032	.4203	.6483	1.5425	2.634	3.229	.847	.4481
26900	10.213	.3413	2.929	234.84	597.18	.8151	.9028	.4188	.6472	1.5451	2.644	3.245		.4465

H (FEET)	P_a ("H_g)	δ	$1/\delta$	T_a (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$\frac{1}{\delta\sqrt{\theta}}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
27000	10.156	.3396	2.942	234.68	596.92	.8144	.9024	.4173	.6460	1.5480	2.655	3.260	.846	.4449
27100	10.112	.3381	2.955	234.48	596.65	.8137	.9020	.4157	.6448	1.5509	2.666	3.276		.4434
27200	10.078	.3368	2.968	234.27	596.45	.8130	.9017	.4143	.6437	1.5535	2.677	3.292	.845	.4419
27300	10.033	.3353	2.982	234.07	596.19	.8123	.9013	.4128	.6425	1.5564	2.687	3.308		.4399
27400	9.988	.3336	2.995	233.87	595.93	.8116	.9009	.4113	.6413	1.5593	2.696	3.325	.844	.4379
27500	9.944	.3323	3.008	233.67	595.66	.8109	.9005	.4098	.6402	1.5620	2.709	3.341		.4367
27600	9.900	.3308	3.022	233.47	595.40	.8102	.9001	.4083	.6390	1.5649	2.720	3.357	.842	.4356
27700	9.856	.3294	3.035	233.29	595.20	.8096	.8998	.4068	.6379	1.5676	2.731	3.374		.4344
27800	9.812	.3279	3.049	233.09	594.93	.8089	.8994	.4054	.6367	1.5706	2.742	3.390	.841	.4333
27900	9.768	.3264	3.063	232.89	594.67	.8082	.8990	.4039	.6356	1.5733	2.753	3.407		.4319
28000	9.724	.3250	3.076	232.69	594.40	.8075	.8986	.4025	.6344	1.5763	2.764	3.423	.840	.4304
28100	9.681	.3235	3.090	232.49	594.14	.8068	.8982	.4010	.6333	1.5790	2.776	3.440		.4290
28200	9.638	.3221	3.104	232.29	593.88	.8061	.8978	.3996	.6321	1.5820	2.787	3.457	.839	.4275
28300	9.595	.3206	3.118	232.08	593.68	.8054	.8975	.3981	.6310	1.5848	2.798	3.474		.4262
28400	9.552	.3192	3.132	231.88	593.41	.8047	.8971	.3967	.6298	1.5878	2.810	3.491	.837	.4250
28500	9.509	.3178	3.146	231.71	593.15	.8041	.8967	.3952	.6287	1.5906	2.821	3.509		.4235
28600	9.466	.3163	3.160	231.51	592.88	.8034	.8963	.3938	.6276	1.5934	2.833	3.526	.836	.4220
28700	9.424	.3149	3.175	231.31	592.62	.8027	.8959	.3923	.6264	1.5964	2.844	3.543		.4205
28800	9.381	.3135	3.189	231.10	592.35	.8020	.8955	.3909	.6253	1.5992	2.856	3.561	.835	.4190
28900	9.339	.3121	3.203	230.90	592.15	.8013	.8952	.3895	.6241	1.6023	2.867	3.579		.4176
29000	9.297	.3107	3.218	230.70	591.89	.8006	.8948	.3881	.6230	1.6051	2.879	3.596	.834	.4162
29100	9.255	.3093	3.232	230.50	591.62	.7999	.8944	.3867	.6219	1.6080	2.891	3.614		.4147
29200	9.213	.3079	3.247	230.30	591.35	.7992	.8940	.3853	.6207	1.6111	2.903	3.632	.833	.4132
29300	9.172	.3065	3.262	230.12	591.09	.7986	.8936	.3838	.6196	1.6139	2.915	3.650		.4119
29400	9.130	.3051	3.276	229.92	590.82	.7979	.8932	.3824	.6184	1.6171	2.927	3.668	.832	.4105
29500	9.089	.3037	3.291	229.72	590.56	.7972	.8928	.3810	.6173	1.6200	2.939	3.686		.4093
29600	9.048	.3024	3.306	229.52	590.36	.7965	.8925	.3796	.6162	1.6228	2.951	3.705	.830	.4081
29700	9.007	.3010	3.321	229.32	590.10	.7958	.8921	.3782	.6150	1.6250	2.963	3.723		.4067
29800	8.966	.2996	3.337	229.12	589.8[illegible]	.7951	.8917	.3769	.6139	1.6289	2.975	3.742	.829	.4053
29900	8.925	.2983	3.352	228.91	589.57	.7944	.8913	.3755	.6128	1.6319	2.987	3.761		.4039
30000	8.885	.2969	3.367	228.71	589.30	.7937	.8909	.3741	.6117	1.6348	3.000	3.779		
30100	8.845	.2956	3.382	228.54	589.05	.7931	.8905	.3727	.6105	1.6380	3.012	3.798	.828	.4025
30200	8.804	.2942	3.398	228.34	588.78	.7924	.8901	.3713	.6094	1.6410	3.024	3.817		.4013
30300	8.764	.2929	3.413	228.14	588.54	.7917	.8898	.3700	.6083	1.6439	3.037	3.837	.826	.4001
30400	8.726	.2915	3.429	227.93	588.32	.7910	.8894	.3686	.6072	1.6469	3.050	3.856		.3987
30500	8.685	.2902	3.445	227.73	588.06	.7903	.8890	.3672	.6060	1.6502	3.062	3.875	.825	.3973
30600	8.645	.2889	3.460	227.53	587.79	.7896	.8886	.3659	.6049	1.6532	3.075	3.894		.3959
30700	8.605	.2876	3.476	227.33	587.53	.7889	.8882	.3645	.6038	1.6562	3.088	3.91[illegible]	.824	.3945
30800	8.566	.2863	3.492	227.13	587.26	.7882	.8878	.3632	.6027	1.6592	3.100	3.914		.3932
30900	8.527	.2849	3.508	226.95	587.00	.7876	.8874	.3618	.6016	1.6622	3.113	3.951	.823	.3918

H (FEET)	P_m ("Hg)	δ	$1/\delta$	T_s (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ψ	$\frac{\delta}{\psi\sqrt{\theta}}$
31000	8.484	.2836	3.524	226.75	586.80	.7869	.8871	.3605	.6004	1.6656	3.127	3.974	.822	.3690
31100	8.449	.2823	3.541	226.55	586.53	.7862	.8867	.3592	.5993	1.6686	3.140	3.994		.3676
31200	8.410	.2811	3.557	226.35	586.27	.7855	.8863	.3578	.5982	1.6717	3.153	4.014	.821	.3663
31300	8.3[illegible]	.2798	3.573	226.15	586.01	.7848	.8859	.3565	.5971	1.6748	3.166	4.034		.3650
31400	8.3[illegible]	.2785	3.590	225.95	585.74	.7841	.8855	.3551	.5960	1.6779	3.179	4.054	.819	.3638
31500	8.2[illegible]5	.2772	3.607	225.74	585.48	.7834	.8851	.3538	.5949	1.6810	3.192	4.075		.3625
31600	8.2[illegible]	.2759	3.623	225.54	585.21	.7827	.8847	.3525	.5938	1.6841	3.205	4.095	.818	.3613
31700	8.218	.2746	3.640	225.37	584.95	.7821	.8843	.3512	.5926	1.6873	3.219	4.116		.3599
31800	8.181	.2734	3.657	225.17	584.68	.7814	.8839	.3499	.5916	1.6905	3.232	4.137	.817	.3586
31900	8.143	.27[illegible]	3.674	224.97	584.4[illegible]	.7807	.8835	.3486	.5904	1.6938	3.246	4.158		.3574
32000	8.105	.2709	3.691	224.76	584.22	.7800	.8832	.3473	.5893	1.6969	3.260	4.180	.815	.3763
32100	8.068	.2696	3.708	224.56	583.95	.7793	.8828	.3460	.5882	1.7001	3.274	4.201		.3750
32200	8.030	.2684	3.725	[illegible]	583.69	.7786	.8824	.3447	.5871	1.7033	3.287	4.222	.814	.3736
32300	7.994	.2671	3.74[illegible]	224.16	583.43	.7779	.8820	.3434	.5860	1.7065	3.301	4.244		.3721
32400	7.956	.2659	3.760	223.96	583.16	.7772	.8816	.3421	.5849	1.7097	3.315	4.265	.813	.3706
32500	7.919	.2646	3.778	223.79	582.90	.7765	.8812	.3408	.5838	1.7129	3.329	4.287		.3693
32600	7.882	.2634	3.795	223.58	582.63	.7759	.8808	.3395	.5827	1.7161	3.343	4.309	.812	.3681
32700	7.846	.2622	3.813	223.38	582.37	.7752	.8804	.3382	.5816	1.7194	3.357	4.331		.3667
32800	7.809	.2610	3.831	223.18	582.17	.7745	.8801	.3370	.5805	1.7227	3.371	4.353	.811	.3655
32900	7.773	.2597	3.849	222.98	581.91	.7738	.8797	.3357	.5794	1.7259	3.386	4.376		.3644
33000	7.737	.2585	3.867	222.78	581.64	.7731	.8793	.3344	.5783	1.7292	3.400	4.398	.809	.3634
33100	7.700	.2573	3.885	222.57	581.38	.7724	.8789	.3332	.5772	1.7325	3.414	4.421		.3620
33200	7.665	.2561	3.903	222.37	581.11	.7717	.8785	.3319	.5764	1.7355	3.429	4.443	.808	.3607
33300	7.629	.2549	3.922	222.20	580.85	.7711	.8781	.3306	.5751	1.7388	3.443	4.466		.3595
33400	7.593	.2537	3.940	222.00	580.58	.7704	.8777	.3294	.5760	1.7422	3.458	4.489	.807	.3582
33500	7.557	.2525	3.958	221.80	580.32	.7697	.8773	.3281	.5729	1.7455	3.473	4.512		.3568
33600	7.522	.2514	3.977	221.60	580.05	.7690	.8769	.3269	.5718	1.7489	3.487	4.535	.806	.3555
33700	7.487	.2502	3.996	221.39	579.79	.7683	.8765	.3256	.5707	1.7522	3.502	4.559		.3543
33800	7.452	.2490	4.015	221.19	579.52	.7676	.8761	.3244	.5696	1.7556	3.517	4.582	.805	.3530
33900	7.417	.2478	4.034	220.99	579.26	.7669	.8757	.3232	.5685	1.7590	3.532	4.606		.3519
34000	7.382	.2467	4.053	220.79	579.06	.7662	.8754	.3219	.5674	1.7624	3.546	4.630	.803	.3508
34100	7.347	.2455	4.072	220.62	578.80	.7656	.8750	.3207	.5664	1.7655	2.563	4.654		.3496
34200	7.312	.2444	4.091	220.41	578.53	.7649	.8746	.3195	.5653	1.7690	3.578	4.678	.802	.3484
34300	7.278	.2432	4.110	220.21	578.27	.7642	.8742	.3185	.5642	1.7724	3.593	4.703		.3471
34400	7.244	.2421	4.130	220.01	578.00	.7635	.8738	.3171	.5631	1.7759	3.609	4.727	.801	.3458
34500	7.209	.2409	4.150	219.81	577.74	.7628	.8734	.3158	.5620	1.7794	3.624	4.751		.3435
34600	7.175	.2398	4.169	219.61	577.47	.7621	.8730	.3146	.5610	1.7825	3.640	4.776	.800	.3432
34700	7.141	.2386	4.189	219.41	577.21	.7614	.8726	.3134	.5599	1.7860	3.655	4.801		.3422
34800	7.107	.2375	4.209	219.26	576.95	.7607	.8722	.3122	.5588	1.7895	3.671	4.826	.799	.3412
34900	7.074	.2364	4.229	219.01	576.66	.7601	.8718	.3110	.5577	1.7931	3.687	4.851		.3400

H (FEET)	P_B (IN.)	δ	$1/\delta$	T_a (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
35000	7.040	.2353	4.249	218.83	576.42	.7594	.8714	.3098	.5567	1.7963	3.703	4.877	.797	.3387
35100	7.007	.2341	4.270	218.63	576.15	.7587	.8710	.3086	.5556	1.7999	3.719	4.902		.3375
35200	6.973	.2330	4.290	218.43	575.89	.7580	.8706	.3075	.5545	1.8034	3.735	4.928	.796	.3363
35300	6.940	.2319	4.311	218.22	575.62	.7573	.8702	.3063	.5534	1.8070	3.751	4.953		.3350
35400	6.907	.2308	4.331	218.02	575.36	.7566	.8698	.3051	.5524	1.8103	3.767	4.979	.795	.3338
35500	6.874	.2297	4.352	217.82	575.09	.7559	.8694	.3039	.5513	1.8139	3.783	5.005		.3326
35600	6.841	.2286	4.373	217.62	574.83	.7552	.8690	.3027	.5502	1.8175	3.800	5.032	.794	.3314
35700	6.809	.2275	4.394	217.45	574.57	.7546	.8686	.3016	.5492	1.8208	3.816	5.058		.3304
35800	6.776	.2264	4.415	217.24	574.37	.7539	.8683	.3004	.5481	1.8245	3.833	5.085	.792	.3293
35900	6.744	.2254	4.436	217.04	574.10	.7532	.8679	.2992	.5471	1.8278	3.850	5.112		.3281
36000	6.711	.2243	4.457	216.84	573.84	.7525	.8675	.2981	.5460	1.8315	3.867	5.139	.791	.3269
36100	6.679	.2232	4.479	216.66	573.58	.7519	.8671	.2969	.5449	1.8352	3.884	5.166		.3256
36200	6.647	.2221	4.501	216.66	573.58	.7519	.8671	.2954	.5436	1.8396	3.902	5.190	.790	.3243
36300	6.615	.2211	4.522	216.66	573.58	.7519	.8671	.2940	.5423	1.8441	3.921	5.215		.3227
36400	6.584	.2200	4.544	216.66	573.58	.7519	.8671	.2926	.5410	1.8485	3.940	5.240	.790	.3211
36500	6.552	.2189	4.566	216.66	573.58	.7519	.8671	.2912	.5397	1.8530	3.959	5.266		.3196
36600	6.521	.2179	4.588	216.66	573.58	.7519	.8671	.2898	.5384	1.8574	3.978	5.291	.790	.3181
36700	6.489	.2168	4.610	216.66	573.58	.7519	.8671	.2884	.5371	1.8619	3.997	5.317		.3167
36800	6.458	.2158	4.632	216.66	573.58	.7519	.8671	.2870	.5358	1.8664	4.017	5.342	.790	.3152
36900	6.427	.2148	4.655	216.66	573.58	.7519	.8671	.2857	.5345	1.8709	4.036	5.366		.3137
37000	6.396	.2137	4.677	216.66	573.58	.7519	.8671	.2843	.5332	1.8753	4.055	5.394	.790	.3121
37100	6.365	.2127	4.700	216.66	573.58	.7519	.8671	.2829	.5319	1.8799	4.075	5.420		.3105
37200	6.335	.2117	4.722	216.66	573.58	.7519	.8671	.2816	.5307	1.8844	4.095	5.446	.790	.3090
37300	6.305	.2107	4.745	216.66	573.58	.7519	.8671	.2802	.5294	1.8889	4.114	5.472		.3076
37400	6.274	.2097	4.768	216.66	573.58	.7519	.8671	.2789	.5281	1.8935	4.134	5.499	.790	.3061
37500	6.244	.2087	4.791	216.66	573.58	.7519	.8671	.2775	.5269	1.8980	4.154	5.525		.3047
37600	6.215	.2077	4.814	216.66	573.58	.7519	.8671	.2762	.5256	1.9026	4.174	5.552	.790	.3032
37700	6.185	.2067	4.837	216.66	573.58	.7519	.8671	.2749	.5243	1.9072	4.194	5.579		.3017
37800	6.155	.2057	4.861	216.66	573.58	.7519	.8671	.2736	.5231	1.9117	4.215	5.605	.790	.3002
37900	6.125	.2047	4.884	216.66	573.58	.7519	.8671	.2723	.5218	1.9164	4.235	5.632		.2988
38000	6.096	.2037	4.908	216.66	573.58	.7519	.8671	.2709	.5206	1.9210	4.255	5.660	.790	.2975
38100	6.067	.2027	4.932	216.66	573.58	.7519	.8671	.2697	.5193	1.9256	4.276	5.687		[illegible]
38200	6.038	.20160	4.955	216.66	573.58	.7519	.8671	.2684	.5181	1.9302	4.296	5.714	.790	.2945
38300	6.009	.20083	4.979	216.66	573.58	.7519	.8671	.2671	.5169	1.9349	4.317	5.742		.2932
38400	5.980	.19987	5.003	216.66	573.58	.7519	.8671	.2658	.5156	1.9395	4.338	5.770	.790	.2918
38500	5.951	.19892	5.027	216.66	573.58	.7519	.8671	.2645	.5144	1.9442	4.359	5.797		.2904
38600	5.923	.19797	5.051	216.66	573.58	.7519	.8671	.2633	.5131	1.9488	4.380	5.825	.790	.2890
38700	5.894	.19701	5.075	216.66	573.58	.7519	.8671	.2620	.5119	1.9535	4.401	5.853		.2876
38800	5.866	.19607	5.100	216.66	573.58	.7519	.8671	.2607	.5107	1.9583	4.422	5.881	.790	.2862
38900	5.838	.19513	5.124	216.66	573.58	.7519	.8671	.2595	.5094	1.9630	4.443	5.910		.2849

h (FEET)	P_s (IN. Hg)	δ	1/δ	T_s (°K)	a (KNOTS)	θ	√θ	σ	√σ	1/√σ	√θ/δ	(1/δ)√θ	φ	δ/(φ√θ)
39000	5.810	.19419	5.149	216.66	573.58	.7519	.8671	.2582	.5082	1.9677	4.465	5.938	.790	.2833
39100	5.782	.19326	5.174	216.66	573.58	.7519	.8671	.2570	.5070	1.9724	4.486	5.967		.2821
39200	5.754	.19233	5.199	216.66	573.58	.7519	.8671	.2558	.5058	1.9772	4.508	5.996	.790	.2807
39300	5.727	.19142	5.224	216.66	573.58	.7519	.8671	.2545	.5046	1.9819	4.529	6.024		.2794
39400	[illegible]	.19049	5.249	216.66	573.58	.7519	.8671	.2533	.5033	1.9867	4.551	6.054	.790	.2781
39500	[illegible]	.18958	5.274	216.66	573.58	.7519	.8671	.2521	.5021	1.9915	4.573	6.083		.2767
39600	[illegible]	.18868	5.300	216.66	573.58	.7519	.8671	.2509	.5009	[illegible]	4.595	6.112	.790	.2754
39700	[illegible]	[illegible]	5.325	216.66	573.58	.7519	.8671	.2497	.4997	[illegible]	4.618	6.142		.2741
39800	[illegible]	[illegible]	5.351	216.66	573.58	.7519	.8671	.2485	.4985	[illegible]	4.640	6.171	.790	.2728
39900	[illegible]	[illegible]	[illegible]	216.66	573.58	.7519	.8671	.2473	.4973	[illegible]	4.662	6.201		.2715
40000	[illegible]	[illegible]	5.403	216.66	573.58	.7519	.8671	.2461	.4961	[illegible]	4.685	6.231	.790	.2702
40100	[illegible]	[illegible]	5.429	216.66	573.58	.7519	.8671	.2449	.4949	[illegible]	4.707	6.261		.2688
40200	[illegible]	[illegible]	5.455	216.66	573.58	.7519	.8671	.2438	.4938	[illegible]	4.730	6.291	.790	.2675
40300	[illegible]	[illegible]	5.481	216.66	573.58	.7519	.8671	.2426	.4926	[illegible]	4.753	6.321		.2663
40400	[illegible]	[illegible]	5.508	216.66	573.58	.7519	.8671	.2414	.4914	[illegible]	4.776	6.352	.790	.2651
40500	[illegible]	[illegible]	5.534	216.66	573.58	.7519	.8671	.2403	.4902	[illegible]	4.799	6.382		.2637
40600	[illegible]	.17482	5.561	216.66	573.58	.7519	.8671	.2391	.4890	2.0448	4.822	6.413	.790	.2624
40700	[illegible]	[illegible]	5.587	216.66	573.58	.7519	.8671	.2380	.4879	2.0497	4.845	6.444		.2612
40800	[illegible]	[illegible]	5.614	216.66	573.58	.7519	.8671	.2368	.4867	[illegible]	4.868	6.475	.790	.2600
40900	[illegible]	[illegible]	5.642	216.66	573.58	.7519	.8671	.2357	.4855	[illegible]	4.892	6.506		.2588
41000	[illegible]	[illegible]	5.668	216.66	573.58	.7519	.8671	.2346	.4844	[illegible]	4.915	6.537	.790	.2575
41100	[illegible]	[illegible]	5.696	216.66	573.58	.7519	.8671	.2334	.4832	[illegible]	4.939	[illegible]		.2560
41200	[illegible]	.17471	5.723	216.66	573.58	.7519	.8671	.2323	.4820	[illegible]	4.963	6.600	.790	.2546
41300	5.202	.17387	5.751	216.66	573.58	.7519	.8671	.2312	.4808	2.0795	4.987	6.632		.2535
41400	5.177	.17303	5.779	216.66	573.58	.7519	.8671	.2301	.4797	2.0845	5.011	6.664	.790	.2525
41500	5.152	.17221	5.806	216.66	573.58	.7519	.8671	.2290	.4786	2.0895	5.035	6.696		.2514
41600	5.127	.17138	5.835	216.66	573.58	.7519	.8671	.2279	.4774	2.0946	5.059	6.729	.790	.2502
41700	5.103	.17056	5.863	216.66	573.58	.7519	.8671	.2268	.4763	2.0996	5.083	6.761		.2490
41800	5.079	.16974	5.891	216.66	573.58	.7519	.8671	.2257	.4751	2.1046	5.108	6.794	.790	.2477
41900	5.054	.16893	5.919	216.66	573.58	.7519	.8671	.2245	.4740	2.1097	5.132	6.826		.2465
42000	5.030	.16812	5.948	216.66	573.58	.7519	.8671	.2236	.4729	2.1148	5.157	6.859	.790	.2452
42100	5.006	.16731	5.976	216.66	573.58	.7519	.8671	.2225	.4717	2.1199	5.182	6.892		.2441
42200	4.982	.16651	6.005	216.66	573.58	.7519	.8671	.2214	.4706	2.1250	5.207	6.926	.790	.2430
42300	4.958	.16571	6.034	216.66	573.58	.7519	.8671	.2204	.4695	2.1301	5.232	6.959		.2419
42400	4.934	.16492	6.063	216.66	573.58	.7519	.8671	.2193	.4683	2.1352	5.257	6.992	.790	.2407
42500	4.910	.16412	6.093	216.66	573.58	.7519	.8671	.2182	.4672	2.1404	5.283	7.026		.2395
42600	4.887	.16334	6.122	216.66	573.58	.7519	.8671	.2172	.4661	2.1455	5.308	7.060	.790	.2383
42700	4.863	.16255	6.152	216.66	573.58	.7519	.8671	.2162	.4650	2.1507	5.334	7.094		.2372
42800	4.840	.16178	6.181	216.66	573.58	.7519	.8671	.2151	.4639	2.1558	5.359	7.128	.790	.2360
42900	4.817	.16100	6.211	216.66	573.58	.7519	.8671	.2141	.4627	2.1610	5.385	7.163		.2349

H (FEET)	P_a ("H_g)	δ	1/δ	T_a (°K)	a (KNOTS)	θ	√θ	σ	√σ	1/√σ	√θ/δ	1/δ√θ	φ	δ/(φ√θ)
43000	4.794	.16023	6.241	216.66	573.58	.7519	.8671	.2131	.4616	2.1662	5.411	7.197	.790	.2338
43100	4.771	.15946	6.271	216.66	573.58	.7519	.8671	.2120	.4605	2.1715	5.437	7.232		.2327
43200	4.748	.15870	6.301	216.66	573.58	.7519	.8671	.2110	.4594	2.1767	5.463	7.266	.790	.2316
43300	4.725	.15794	6.331	216.66	573.58	.7519	.8671	.2100	.4583	2.1819	5.490	7.301		.2305
43400	4.702	.15718	6.362	216.66	573.58	.7519	.8671	.2090	.4572	2.1872	5.516	7.337	.790	.2294
43500	4.680	.15642	6.393	216.66	573.58	.7519	.8671	.2080	.4561	2.1924	5.543	7.372		.2283
43600	4.657	.15567	6.423	216.66	573.58	.7519	.8671	.2070	.4550	2.1977	5.570	7.408	.790	.2272
43700	4.635	.15492	6.454	216.66	573.58	.7519	.8671	.2060	.4539	2.2030	5.597	7.444		.2261
43800	4.613	.15418	6.485	216.66	573.58	.7519	.8671	.2050	.4528	2.2083	5.623	7.479	.790	.2251
43900	4.591	.15345	6.516	216.66	573.58	.7519	.8671	.2040	.4518	2.2136	5.650	7.515		.2240
44000	4.569	.15271	6.548	216.66	573.58	.7519	.8671	.20310	.4507	2.2189	5.678	7.551	.790	.2229
44100	4.547	.15198	6.579	216.66	573.58	.7519	.8671	.20213	.4496	2.2242	5.705	7.588		.2218
44200	4.525	.15125	6.611	216.66	573.58	.7519	.8671	.20117	.4485	2.2296	5.732	7.624	.790	.2208
44300	4.503	.15053	6.643	216.66	573.58	.7519	.8671	.20020	.4474	2.2349	5.760	7.661		.2197
44400	4.482	.14980	6.675	216.66	573.58	.7519	.8671	.19924	.4464	2.2403	5.788	7.698	.790	.2186
44500	4.460	.14908	6.707	216.66	573.58	.7519	.8671	.19828	.4453	2.2457	5.816	7.735		.2176
44600	4.439	.14837	6.739	216.66	573.58	.7519	.8671	.19733	.4442	2.2511	5.844	7.772	.790	.2166
44700	4.418	.14766	6.772	216.66	573.58	.7519	.8671	.19639	.4432	2.2566	5.872	7.810		.2155
44800	4.397	.14695	6.805	216.66	573.58	.7519	.8671	.19545	.4421	2.2620	5.900	7.847	.790	.2145
44900	4.375	.14624	6.838	216.66	573.58	.7519	.8671	.19451	.4410	2.2674	5.929	7.886		.2134
45000	4.354	.14554	6.871	216.66	573.58	.7519	.8671	.19358	.4400	2.2729	5.957	7.923	.790	.2124
45100	4.334	.14485	6.903	216.66	573.58	.7519	.8671	.19265	.4389	2.2783	5.986	7.961		.2115
45200	4.313	.14415	6.937	216.66	573.58	.7519	.8671	.19173	.4379	2.2838	6.015	8.000	.790	.2105
45300	4.292	.14346	6.970	216.66	573.58	.7519	.8671	.19080	.4368	2.2893	6.044	8.038		.2095
45400	4.272	.14277	7.004	216.66	573.58	.7519	.8671	.18989	.4358	2.2948	6.073	8.077	.790	.2084
45500	4.251	.14208	7.038	216.66	573.58	.7519	.8671	.18897	.4347	2.3004	6.102	8.116		.2074
45600	4.231	.14141	7.071	216.66	573.58	.7519	.8671	.18807	.4337	2.3059	6.131	8.155	.790	.2064
45700	4.210	.14073	7.105	216.66	573.58	.7519	.8671	.18717	.4326	2.3114	6.161	8.194		.2054
45800	4.190	.14005	7.140	216.66	573.58	.7519	.8671	.18627	.4316	2.3170	6.191	8.234	.790	.2045
45900	4.170	.13938	7.174	216.66	573.58	.7519	.8671	.18538	.4306	2.3226	6.221	8.274		.2034
46000	4.150	.13871	7.209	216.66	573.58	.7519	.8671	.18449	.4295	2.3282	6.251	8.314	.790	.2024
46100	4.130	.13805	7.243	216.66	573.58	.7519	.8671	.18360	.4285	2.3338	6.281	8.353		.2015
46200	4.110	.13739	7.278	216.66	573.58	.7519	.8671	.18273	.4275	2.3394	6.311	8.394	.790	.2005
46300	4.091	.13672	7.314	216.66	573.58	.7519	.8671	.18184	.4264	2.3450	6.342	8.435		.1995
46400	4.071	.13607	7.349	216.66	573.58	.7519	.8671	.18098	.4254	2.3507	6.372	8.475	.790	.1986
46500	4.051	.13542	7.384	216.66	573.58	.7519	.8671	.18011	.4244	2.3563	6.403	8.516		.1977
46600	4.032	.13477	7.420	216.66	573.58	.7519	.8671	.17924	.4234	2.3620	6.433	8.557	.790	.1967
46700	4.013	.13412	7.456	216.66	573.58	.7519	.8671	.17839	.4224	2.3677	6.465	8.598		.1957
46800	3.993	.13348	7.491	216.66	573.58	.7519	.8671	.17753	.4213	2.3734	6.496	8.639	.790	.1948
46900	3.974	.13284	7.527	216.66	573.58	.7519	.8671	.17668	.4203	2.3791	6.527	8.681		.1938

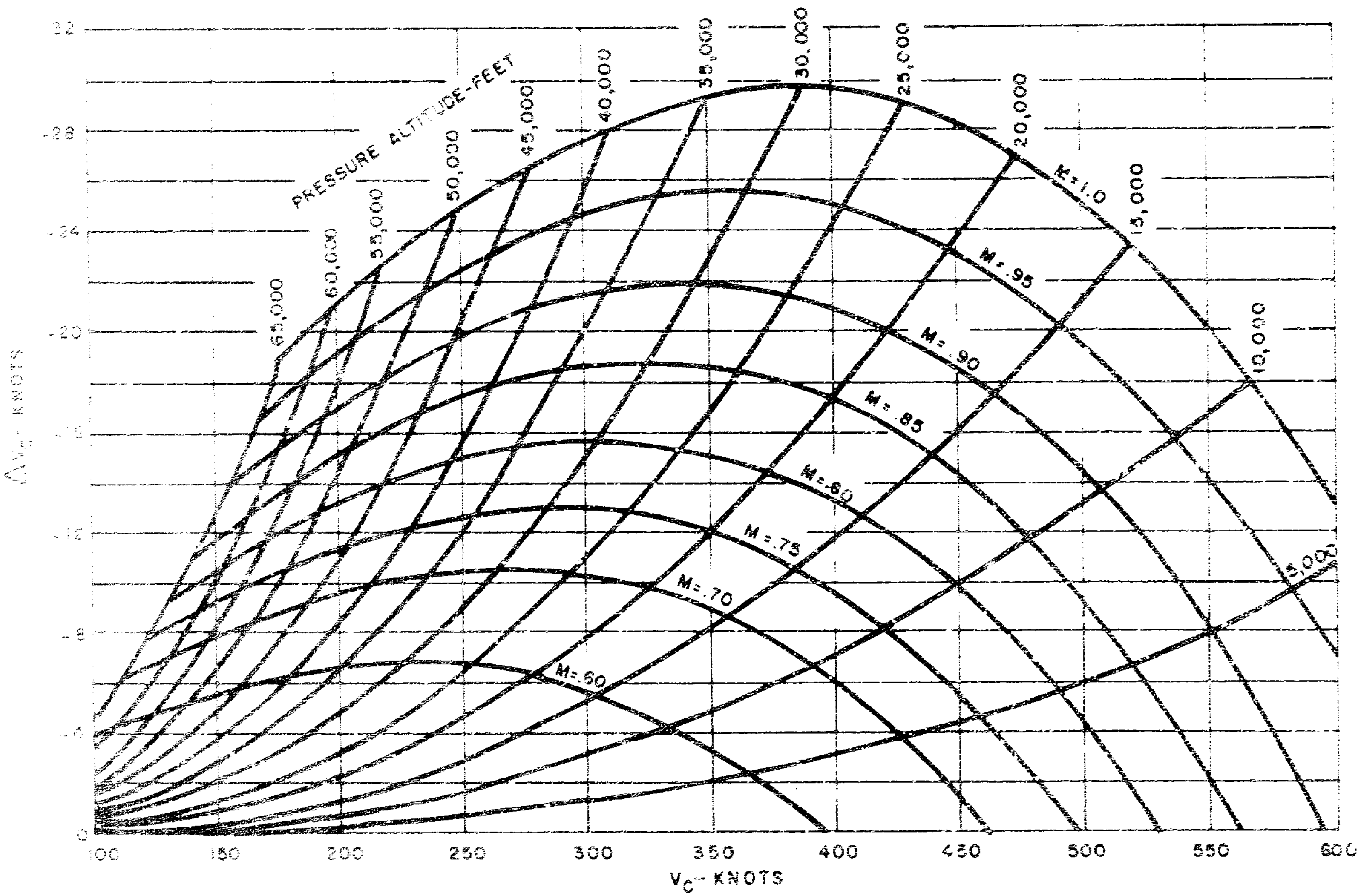
H (FEET)	P (IN. Hg)	δ	$1/\delta$	T_a (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
47000	3.955	.13221	7.563	216.66	573.58	.7519	.8671	.17584	.4193	2.3848	6.558	8.722	.790	.1929
47100	3.936	.13157	7.600	216.66	573.58	.7519	.8671	.17499	.4183	2.3905	6.590	8.765		.1919
47200	3.917	.13094	7.637	216.66	573.58	.7519	.8671	.17415	.4173	2.3963	6.622	8.807	.790	.1910
47300	3.899	.13031	7.674	216.66	573.58	.7519	.8671	.17331	.4163	2.4021	6.654	8.850		.1502
47400	3.880	.12969	7.710	216.66	573.58	.7519	.8671	.17249	.4153	2.4073	6.685	8.892	.790	.1893
47500	3.861	.12907	7.747	216.66	573.58	.7519	.8671	.17166	.4143	2.4136	6.718	8.935		.1884
47600	3.843	.12845	7.785	216.66	573.58	.7519	.8671	.17084	.4133	2.4194	6.750	8.978	.790	.1875
47700	3.824	.12783	7.822	216.66	573.58	.7519	.8671	.17002	.4123	2.4252	6.783	9.021		.1665
47800	3.806	.12722	7.860	216.66	573.58	.7519	.8671	.16920	.4113	2.4311	6.815	9.065	.790	.1856
47900	3.788	.12660	7.898	216.66	573.58	.7519	.8671	.16839	.4103	2.4373	6.849	9.109		.1848
48000	3.770	.12600	7.936	216.66	573.58	.7519	.8671	.16758	.4094	2.4428	6.881	9.152	.790	.1839
48100	3.752	.12540	7.974	216.66	573.58	.7519	.8671	.16678	.4084	2.4487	6.914	9.196		.1830
48200	3.734	.12480	8.012	216.66	573.58	.7519	.8671	.16598	.4074	2.4546	6.947	9.240	.790	.1821
48300	3.716	.12419	8.052	216.66	573.58	.7519	.8671	.16518	.4064	2.4605	6.982	9.286		.1813
48400	3.698	.12360	8.090	216.66	573.58	.7519	.8671	.16439	.4054	2.4664	7.015	9.330	.790	.1804
48500	3.680	.12301	8.129	216.66	573.58	.7519	.8671	.16360	.4045	2.4723	7.049	9.375		.1795
48600	3.662	.12242	8.168	216.66	573.58	.7519	.8671	.16282	.4035	2.4783	7.083	9.420	.790	.1786
48700	3.645	.12183	8.208	216.66	573.58	.7519	.8671	.16204	.4025	2.4842	7.117	9.466		.1778
48800	3.627	.12125	8.247	216.66	573.58	.7519	.8671	.16126	.4016	2.4902	7.151	9.511	.790	.1770
48900	3.610	.12067	8.287	216.66	573.58	.7519	.8671	.16049	.4006	2.4962	7.185	9.557		.1762
49000	3.593	.12009	8.327	216.66	573.58	.7519	.8671	.15972	.3996	2.5022	7.220	9.603	.790	.1753
49100	3.576	.11951	8.367	216.66	573.58	.7519	.8671	.15895	.3987	2.5082	7.255	9.649		.1744
49200	3.558	.11894	8.407	216.66	573.58	.7519	.8671	.15819	.3977	2.5145	7.290	9.696	.790	.1735
49300	3.541	.11837	8.448	216.66	573.58	.7519	.8671	.15743	.3968	2.5203	7.325	9.742		.1727
49400	3.524	.11780	8.489	216.66	573.58	.7519	.8671	.15665	.3958	2.5264	7.360	9.789	.790	.1719
49500	3.507	.11724	8.529	216.66	573.58	.7519	.8671	.15593	.3949	2.5325	7.395	9.836		.1711
49600	3.491	.11668	8.570	216.66	573.58	.7519	.8671	.15518	.3939	2.5385	7.431	9.883	.790	.1703
49700	3.474	.11611	8.612	216.66	573.58	.7519	.8671	.15443	.3930	2.5447	7.467	9.932		.1795
49800	3.457	.11556	8.653	216.66	573.58	.7519	.8671	.15369	.3920	2.5508	7.503	9.979	.790	.1687
49900	3.441	.11500	8.695	216.66	573.58	.7519	.8671	.15295	.3911	2.5570	7.540	10.028		.1675
50000	3.424	.11445	8.737	216.66	573.58	.7519	.8671	.15222	.3902	2.5631	7.576	10.076	.790	.1671
50100	3.408	.11390	8.779	216.66	573.58	.7519	.8671	.15149	.3892	2.5692	7.612	10.125		.1663
50200	3.391	.11336	8.821	216.66	573.58	.7519	.8671	.15077	.3883	2.5754	7.649	10.175	.790	.1655
50300	3.375	.11281	8.864	216.66	573.58	.7519	.8671	.15004	.3874	2.5816	7.686	10.222		.1647
50400	3.359	.11227	8.907	216.66	573.58	.7519	.8671	.14932	.3864	2.5879	7.723	10.272	.790	.1639
50500	3.343	.11173	8.950	216.66	573.58	.7519	.8671	.14861	.3855	2.5941	7.760	10.321		.1631
50600	3.327	.11120	8.992	216.66	573.58	.7519	.8671	.14789	.3846	2.6003	7.797	10.371	.790	.1623
50700	3.311	.11067	9.035	216.66	573.58	.7519	.8671	.14719	.3837	2.6065	7.833	10.420		.1615
50800	3.295	.11014	9.079	216.66	573.58	.7519	.8671	.14648	.3827	2.6128	7.872	10.470	.790	.1607
50900	3.279	.10961	9.123	216.66	573.58	.7519	.8671	.14578	.3818	2.6191	7.910	10.521		.1600

H FEET	Pa (in Hg)	δ	$1/\delta$	Ta (°K)	a KNOTS	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	ϕ [illegible]
51000	3.263	10908	9.167	216.66	573.58	.7519	.8671	.14508	.3809	2.6254	7.945	10.372	.790	.1592
51100	3.248	10859	9.211	216.66	573.58	.7519	.8671	.14439	.3800	2.6317	7.987	10.623		.1584
51200	3.232	10804	9.255	216.66	573.58	.7519	.8671	.14369	.3791	2.6381	8.025	10.674	.790	.1576
51300	3.217	10752	9.300	216.66	573.58	.7519	.8671	.14300	.3782	2.6444	8.064	10.725		.1569
51400	3.201	10700	9.345	216.66	573.58	.7519	[illegible]	.14231	.3772	2.6508	8.103	10.778	.790	.1562
51500	3.186	10649	9.390	216.66	573.58	.7519	.8671	.14164	.3763	2.6571	8.142	10.829		.1555
51600	3.171	10598	9.435	216.66	573.58	.7519	.8671	.14096	.3754	2.6635	8.181	10.881	.790	.1547
51700	3.155	10547	9.481	216.66	573.58	.7519	.8671	.14028	.3745	2.6700	8.221	10.934		.1540
51800	3.140	10497	9.525	216.66	573.58	.7519	.8671	.13961	.3736	2.6764	8.260	10.986	.790	.1532
51900	3.125	10446	9.573	216.66	573.58	.7519	.8671	.13894	.3727	2.6828	8.300	11.040		.1525
52000	[illegible]	10396	9.619	216.66	573.58	.7519	.8671	.13827	.3719	2.6893	8.340	11.093	.790	.1518
52100	[illegible]	10347	9.664	216.66	573.58	.7519	.8671	.13761	.3710	2.6957	8.380	11.145		.1510
52200	3.080	10297	9.711	216.66	573.58	.7519	.8671	.13695	.3701	2.7022	8.420	11.199	.790	.1503
52300	3.066	10247	9.759	216.66	573.58	.7519	.8671	.13629	.3692	2.7088	8.462	11.254		.1496
52400	3.051	10198	9.805	216.66	573.58	.7519	.8671	.13564	.3683	2.7152	8.502	11.309	.790	.1489
52500	3.036	10149	9.853	216.66	573.58	.7519	.8671	.13499	.3674	2.7218	8.543	11.363		.1482
52600	3.022	10101	9.900	216.66	573.58	.7519	.8671	.13434	.3665	2.7283	8.584	11.417	.790	.1474
52700	3.007	10052	9.948	216.66	573.58	.7519	.8671	.13370	.3656	2.7349	8.626	11.472		.1467
52800	2.993	10004	9.996	216.66	573.58	.7519	.8671	.13306	.3648	2.7415	8.667	11.527	.790	.1459
52900	2.979	09956	10.044	216.66	573.58	.7519	.8671	.13242	.3639	2.7481	8.709	11.583		.1453
53000	2.965	09908	10.092	216.66	573.58	.7519	.8671	.13178	.3630	2.7547	8.751	11.639	.790	.1446
53100	2.950	09861	10.141	216.66	573.58	.7519	.8671	.13115	.3621	2.7613	8.793	11.695		.1439
53200	2.936	09813	10.190	216.66	573.58	.7519	.8671	.13052	.3613	2.7680	8.836	11.752	.790	.1432
53300	2.922	09766	10.239	216.66	573.58	.7519	.8671	.12989	.3604	2.7746	8.878	11.808		.1425
53400	2.908	09719	10.289	216.66	573.58	.7519	.8671	.12927	.3595	2.7813	8.921	11.865	.790	.1418
53500	2.894	09673	10.338	216.66	573.58	.7519	.8671	.12865	.3587	2.7880	8.964	11.922		.1411
53600	2.880	09627	10.387	216.66	573.58	.7519	.8671	.12803	.3578	2.7947	[illegible]	11.979	.790	.1405
53700	2.866	09581	10.437	216.66	573.58	.7519	.8671	.12743	.3570	2.8014	9.050	12.036		.1398
53800	2.852	09535	10.487	216.66	573.58	.7519	.8671	.12681	.3561	2.8082	9.093	12.094	.790	.1391
53900	2.839	09489	10.538	216.66	573.58	.7518	.8671	.12620	.3552	2.8149	9.137	12.153		.1384
54000	2.825	.09443	10.589	216.66	573.58	.7519	.8671	.12560	.3544	2.8217	9.182	12.212		.1378
54100	2.812	.09398	10.640	216.66	573.58	.7519	.8671	.12500	.3536	2.8285	9.226	12.271	.790	.1371
54200	2.798	.09353	10.691	216.66	573.58	.7519	.8671	.12440	.3527	2.8353	9.270	12.330		.1364
54300	2.785	.09308	10.743	216.66	573.58	.7519	.8671	.12380	.3518	2.8421	9.315	12.389	.790	.1357
54400	2.771	.09264	10.794	216.66	573.58	.7519	.8671	.12321	.3510	2.8489	9.359	12.448		.1351
54500	2.758	.09219	10.847	216.66	573.58	.7519	.8671	.12261	.3502	2.8558	9.405	12.509	.790	.1346
54600	2.745	.09175	10.899	216.66	573.58	.7519	.8671	.12203	.3493	2.8627	9.450	12.569		.1340
54700	2.732	.09131	10.951	216.66	573.58	.7519	.8671	.12144	.3485	2.8695	9.496	12.630	.790	.1333
54800	2.719	.09087	11.004	216.66	573.58	.7519	.8671	.12086	.3477	2.8765	9.542	12.691		.1327
54900	2.706	.09044	11.037	216.66	573.58	.7519	.8671	.12028	.3468	2.8834	9.587	12.751	.790	.1320

H (FEET)	P_a ("Hg)	δ	$1/\delta$	Te (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$1/\delta\sqrt{\theta}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
55000	2.693	.09000	11.111	216.66	573.58	.7519	.8671	.11971	.3460	2.8903	9.634	12.813	.790	.1313
55100	2.680	.08957	11.164	216.66	573.58	.7519	.8671	.11913	.3452	2.8973	9.680	12.875		.1304
55200	2.667	.08914	11.218	216.66	573.58	.7519	.8671	.11856	.3443	2.9042	9.727	12.937	.790	.1300
55300	2.654	.08871	11.272	216.66	573.58	.7519	.8671	.11799	.3435	2.9112	9.774	13.000		.1295
55400	2.641	.08829	11.326	216.66	573.58	.7519	.8671	.11743	.3427	2.9182	9.821	13.062	.790	.1289
55500	2.629	.08787	11.380	216.66	573.58	.7519	.8671	.11686	.3419	2.9252	9.867	13.124		.1282
55600	2.616	.08745	11.435	216.66	573.58	.7519	.8671	.11630	.3410	2.9323	9.915	13.187	.790	.1275
55700	2.603	.08702	11.491	216.66	573.58	.7519	.8671	.11574	.3402	2.9394	9.964	13.252		.1270
55800	2.591	.08661	11.546	216.66	573.58	.7519	.8671	.11519	.3394	2.9464	10.011	13.315	.790	.1264
55900	2.578	.08619	11.602	216.66	573.58	.7519	.8671	.11463	.3386	2.9535	10.060	13.380		.1258
56000	2.566	.08578	11.657	216.66	573.58	.7519	.8671	.11409	.3378	2.9606	10.108	13.444	.790	.1252
56100	2.554	.08537	11.713	216.66	573.58	.7519	.8671	.11354	.3370	2.9678	10.156	13.506		.1246
56200	2.542	.08496	11.770	216.66	573.58	.7519	.8671	.11299	.3361	2.9749	10.205	13.577	.790	.1240
56300	2.529	.08455	11.827	216.66	573.58	.7519	.8671	.11245	.3353	2.9820	10.255	13.639		.1233
56400	2.517	.08415	11.883	216.66	573.58	.7519	.8671	.11192	.3345	2.9892	10.304	13.704	.790	.1227
56500	2.505	.08374	11.941	216.66	573.58	.7519	.8671	.11138	.3337	2.9964	10.354	13.771		.1221
56600	2.493	.08334	11.999	216.66	573.58	.7519	.8671	.11085	.3329	3.0036	10.404	13.837	.799	.1216
56700	2.481	.08294	12.056	216.66	573.58	.7519	.8671	.11031	.3321	3.0108	10.454	13.904		.1210
56800	2.469	.08254	12.115	216.66	573.58	.7519	.8671	.10978	.3313	3.0181	10.505	13.972	.790	.1204
56900	2.455	.08215	12.172	216.66	573.58	.7519	.8671	.10926	.3305	3.0253	10.555	14.038		.1199
57000	2.446	.08175	12.232	216.66	573.58	.7519	.8671	.10873	.3297	3.0327	10.606	14.107	.790	.1194
57100	2.434	.08136	12.291	216.66	573.58	.7519	.8671	.10821	.3290	3.0399	10.657	14.174		.1188
57200	2.422	.08097	12.350	216.66	573.58	.7519	.8671	.10769	.3282	3.0472	10.708	14.242	.790	.1182
57300	2.411	.08058	12.410	216.66	573.58	.7519	.8671	.10718	.3274	3.0546	10.760	14.311		.1176
57400	2.399	.08020	12.468	216.66	573.58	.7519	.8671	.10666	.3266	3.0619	10.811	14.379	.790	.1170
57500	2.388	.07981	12.529	216.66	573.58	.7519	.8671	.10615	.3258	3.0693	10.864	14.450		.1165
57600	2.376	.07943	12.589	216.66	573.58	.7519	.8671	.10564	.3250	3.0767	10.916	14.519	.790	.1159
57700	2.365	.07905	12.650	216.66	573.58	.7519	.8671	.10514	.3242	3.0841	10.969	14.588		.1153
57800	2.353	.07867	12.711	216.66	573.58	.7519	.8671	.10463	.3235	3.0915	11.022	14.659	.790	.1148
57900	2.342	.07829	12.773	216.66	573.58	.7519	.8671	.10413	.3227	3.0989	11.075	14.734		.1143
58000	2.331	.07792	12.833	216.66	573.58	.7519	.8671	.10363	.3219	3.1064	11.128	14.800		.1138
58100	2.320	.07754	12.896	216.66	573.58	.7519	.8671	.10313	.3211	3.1139	11.182	14.872	.790	.1133
58200	2.309	.07717	12.958	216.66	573.58	.7519	.8671	.10264	.3204	3.1214	11.236	14.944		.1127
58300	2.293	.07680	13.020	216.66	573.58	.7519	.8671	.10215	.3196	3.1289	11.290	15.016	.790	.1121
58400	2.287	.07643	13.083	216.66	573.58	.7519	.8671	.10166	.3188	3.1364	11.345	15.088		.1115
58500	2.276	.07607	13.145	216.66	573.58	.7519	.8671	.10117	.3181	3.1440	11.398	15.160	.790	.1110
58600	2.265	.07570	13.210	216.66	573.58	.7519	.8671	.10068	.3173	3.1515	11.454	15.234		.1105
58700	2.254	.07534	13.273	216.66	573.58	.7519	.8671	.10020	.3165	3.1591	11.509	15.307	.790	.1100
58800	2.243	.07498	13.336	216.66	573.58	.7519	.8671	.09972	.3158	3.1667	11.564	15.380		.1094
58900	2.234	.07462	13.401	216.66	573.58	.7519	.8671	.09924	.3150	3.1743	11.620	15.454	.790	.1089

H (FEET)	P ("Hg)	δ	1/δ	Ta (°K)	a (KNOTS)	θ	√θ	σ	√σ	1/√σ	√θ/δ	(1/δ)√θ	φ	δ/(φ√θ)
39000	2.221	07426	13.466	216.66	573.58	7519	8671	.09877	3143	3.1020	11.676	15.529	.790	.0894
39100	2.211	07390	13.531	216.66	573.58	7519	8671	.09829	3135	3.1896	11.731	15.605		.0890
39200	2.200	07355	13.596	216.66	573.58	7519	8671	.09782	.3128	3.1973	11.789	15.679	.790	.0886
39300	2.190	07320	13.661	216.66	573.58	7519	8671	.09735	.3120	3.2050	11.845	15.754		.0881
39400	2.179	07285	13.726	216.66	573.58	7519	8671	09689	.3113	3.2127	11.902	15.830	.79	.0877
39500	2.169	07250	13.793	216.66	573.58	7519	8671	.09642	.3105	3.2204	11.960	15.906		.0873
39600	2.158	07215	13.860	216.66	573.58	7519	8671	.09596	.3098	3.2281	12.018	15.984	.790	.0869
39700	2.148	07180	13.927	216.66	573.58	7519	8671	.09550	.3090	3.2359	12.076	16.061		.0865
39800	2.138	07146	13.993	216.66	573.58	7519	8671	.09504	.3083	3.2438	12.134	16.138	.790	.0861
39900	2.127	07112	14.060	216.66	573.58	7519	8671	.09459	.3075	3.2515	12.192	16.215		.0856
40000	2.117	07077	14.130	216.66	573.58	7519	8671	.09413	.3068	3.2594	12.252	16.235	.790	.0852
40100	2.107	07044	14.196	216.66	573.58	7519	8671	.09368	.3061	3.2672	12.303	16.372		.0847
40200	2.097	07010	14.265	216.66	573.58	.7519	8671	.09323	.3053	3.2750	12.369	16.451	.790	.0843
40300	2.087	06976	14.334	216.66	573.58	.7519	8671	09279	.3046	3.2829	12.429	16.531		.0839
40400	2.077	06943	14.403	216.66	573.58	7519	8671	.09234	.3039	3.2908	12.488	16.610	.790	.0836
40500	2.067	06910	14.471	216.66	573.58	7519	8671	09190	.3031	3.2987	12.548	16.689		.0831
40600	2.057	06876	14.543	216.66	573.58	7519	8671	.09146	.3024	3.3067	12.610	16.772	.790	.0827
40700	2.047	06843	14.613	216.66	573.58	7519	8671	.09102	.3017	3.3147	12.671	16.853		.0824
40800	2.037	06810	14.684	216.66	573.58	7519	8671	09058	.3010	3.3226	12.732	16.934	.790	[illegible]
40900	2.028	06778	14.753	216.66	573.58	7519	8671	09015	.3002	3.3306	12.792	17.014		.0816
61000	2.0183	06745	14.825	216.66	573.58	7519	8671	.08972	.2995	3.3386	12.855	17.097		.0813
61100	2.0087	06713	14.896	216.66	573.58	7519	.8671	.08929	.2988	3.3466	12.916	17.179	.790	[illegible]
61200	1.9990	06681	14.967	216.66	573.58	7519	.8671	08886	.2981	3.3547	12.978	17.261		[illegible]
61300	1.9894	.06649	15.039	216.66	573.58	7519	8671	.08843	.2974	3.3628	13.041	17.344	.790	.0800
61400	1.9799	06617	15.112	216.66	573.58	7519	8671	.08801	.2967	3.3709	13.104	17.428		.0797
61500	1.9704	06585	15.186	216.66	573.58	.7519	.8671	.08758	.2959	3.3790	13.167	17.513	.790	.0793
61600	1.9610	06554	15.257	216.66	573.58	.7519	.8671	.08717	.2952	3.3871	13.230	17.596		.0789
61700	1.9515	06522	15.332	216.66	573.58	.7519	.8671	.08675	.2945	3.3953	13.295	17.682	.790	.0785
61800	1.9422	.06491	15.405	216.66	573.58	.7519	.8671	.08633	.2938	3.4035	13.356	17.766		.0782
61900	1.9329	.06460	15.479	216.66	573.58	7519	.8671	.08592	.2931	3.4116	13.422	17.852	.790	.0778
62000	1.9236	06429	15.554	216.66	573.58	.7519	.8671	.08550	.2924	3.4199	13.487	17.938		.0775
62100	1.9144	.06398	15.629	216.66	573.58	7519	8671	.08510	.2917	3.4280	13.552	18.025	.790	.0771
62200	1.9052	06367	15.706	216.66	573.58	.7519	8671	.08469	.2910	3.4363	13.613	18.113		.0767
62300	1.8961	06337	15.780	216.66	573.58	7519	8671	.08428	.2903	3.4445	13.683	18.199	.790	.0763
62400	1.8870	06306	15.857	216.66	573.58	7519	8671	.08388	.2896	3.4529	13.750	18.288		.0759
62500	1.8779	06276	15.933	216.66	573.58	7519	8671	.08347	.2889	3.4612	13.816	18.375	.790	.0755
62600	1.8689	06246	16.010	216.66	573.58	7519	8671	.08307	.2882	3.4695	13.882	18.463		.0752
62700	1.8600	06216	16.087	216.66	573.58	.7519	8671	.08268	.2875	3.4778	13.949	18.552	.790	.0748
62800	1.8510	.06186	16.165	216.66	573.58	.7519	8671	.08228	.2868	3.4862	14.017	18.642		.0745
62900	1.8421	.06157	16.241	216.66	573.58	.7519	8671	.08188	.2862	3.4946	14.083	18.730	.790	.0741

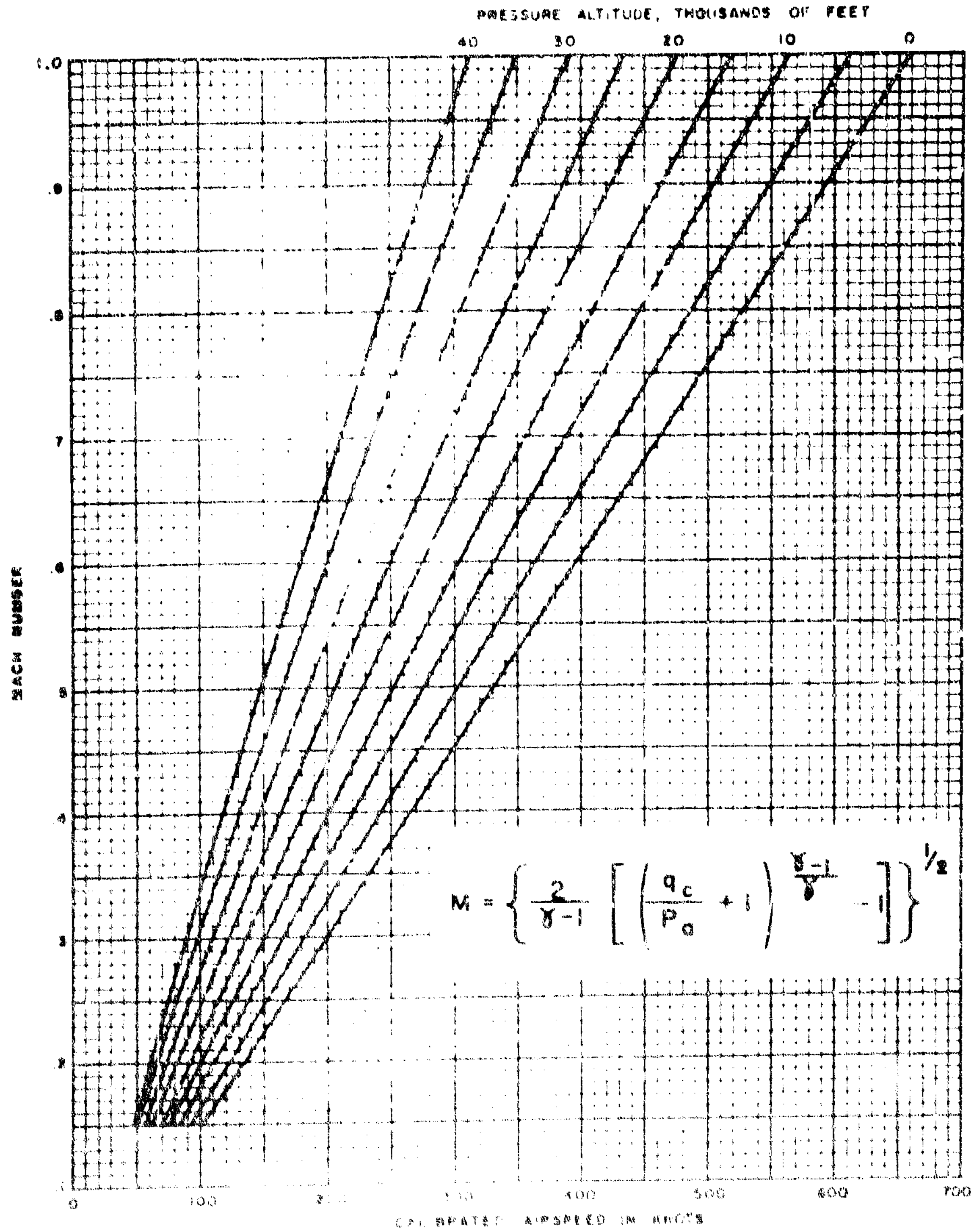
h (FEET)	P_a (″Hg)	δ	$\frac{1}{\delta}$	T_a (°K)	a (KNOTS)	θ	$\sqrt{\theta}$	σ	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\delta}$	$^1/_{\delta\sqrt{\theta}}$	ϕ	$\frac{\delta}{\phi\sqrt{\theta}}$
63000	1.8333	.06127	16.321	216.66	573.58	.7519	.8671	.08149	.2855	3.5030	14.152	18.822	.790	.1084
63100	1.8244	.06098	16.398	216.66	573.58	.7519	.8671	.08110	.2848	3.5114	14.219	18.911		.1079
63200	1.8156	.06068	16.477	216.66	573.58	.7519	.8671	.08071	.2841	3.5199	14.287	19.002	.790	.1074
63300	1.8071	.06039	16.559	216.66	573.58	.7519	.8671	.08032	.2834	3.5284	14.356	19.096		.1068
63400	1.7984	.06010	16.638	216.66	573.58	.7519	.8671	.07994	.2827	3.5369	14.427	19.188	.790	.1063
63500	1.7898	.05982	16.716	216.66	573.58	.7519	.8671	.07956	.2821	3.5453	14.495	19.278		.1058
63600	1.7812	.05953	16.798	216.66	573.58	.7519	.8671	.07918	.2814	3.5539	14.565	19.372	.790	.1054
63700	1.7727	.05924	16.880	216.66	573.58	.7519	.8671	.07880	.2807	3.5624	14.637	19.467		.1048
63800	1.7642	.05896	16.960	216.66	573.58	.7519	.8671	.07842	.2800	3.5710	14.706	19.559	.790	.1043
63900	1.7558	.05868	17.041	216.66	573.58	.7519	.8671	.07804	.2794	3.5796	14.776	19.653		.1038
64000	1.7473	.05840	17.123	216.66	573.58	.7519	.8671	.07767	.2787	3.5882	14.847	19.747	.790	.1033
64100	1.7389	.05812	17.205	216.66	573.58	.7519	.8671	.07730	.2780	3.5968	14.919	19.842		.1028
64200	1.7306	.05784	17.289	216.66	573.58	.7519	.8671	.07693	.2774	3.6055	14.991	19.938	.790	.1023
64300	1.7223	.05756	17.373	216.66	573.58	.7519	.8671	.07656	.2767	3.6141	15.064	20.035		.1018
64400	1.7140	.05728	17.458	216.66	573.58	.7519	.8671	.07619	.2760	3.6229	15.137	20.133	.790	.1013
64500	1.7058	.05701	17.540	216.66	573.58	.7519	.8671	.07582	.2754	3.6316	15.209	20.228		.1011
64600	1.6977	.05674	17.624	216.66	573.58	.7519	.8671	.07546	.2747	3.6403	15.282	20.325	.790	.1004
64700	1.6895	.05647	17.708	216.66	573.58	.7519	.8671	.07510	.2740	3.6490	15.355	20.422		.1001
64800	1.6814	.05620	17.793	216.66	573.58	.7519	.8671	.07474	.2734	3.6578	15.428	20.520	.790	.0994
64900	1.6733	.05592	17.882	216.66	573.58	.7519	.8671	.07438	.2727	3.6667	15.506	20.623		.0988
65000	1.6653	.05566	17.966	216.66	573.58	.7519	.8671	.07402	.2721	3.6755	15.578	20.719	.790	.0983
65100	1.6573	.05539	18.053	216.66	573.58	.7519	.8671	.07367	.2714	3.6843	15.654	20.822		.0979
65200	1.6494	.05513	18.138	216.66	573.58	.7519	.8671	.07332	.2708	3.6932	15.728	20.918	.790	.0975
65300	1.6415	.05486	18.228	216.66	573.58	.7519	.8671	.07296	.2701	3.7021	15.805	21.02		.0971
65400	1.6336	.05460	18.315	216.66	573.58	.7519	.8671	.07261	.2695	3.7110	15.880	21.12	.790	.0966
65500	1.6258	.05433	18.406	216.66	573.58	.7519	.8671	.07227	.2688	3.7199	15.959	21.22		.0961
65600	1.6179	.05407	18.494	216.66	573.58	.7519	.8671	.07192	.2682	3.7289	16.036	21.32	.790	.0956
65700	1.6102	.05382	18.580	216.66	573.58	.7519	.8671	.07158	.2675	3.7378	16.111	21.42		.0952
65800	1.6025	.05356	18.670	216.66	573.58	.7519	.8671	.07123	.2669	3.7468	16.189	21.53	.790	.0947
65900	1.5948	.05330	18.761	216.66	573.58	.7519	.8671	.07089	.2663	3.7558	16.268	21.63		.0943
66000	1.5872	.05305	18.850	216.66	573.58	.7519	.8671	.07055	.2656	3.7648	16.344	21.73	.790	.0938
66100	1.5795	.05279	18.943	216.66	573.58	.7519	.8671	.07021	.2650	3.7740	16.425	21.84		.0934
66200	1.5720	.05254	19.033	216.66	573.58	.7519	.8671	.06987	.2643	3.7830	16.503	21.94	.790	.0929
66300	1.5644	.05229	19.124	216.66	573.58	.7519	.8671	.06954	.2637	3.7921	16.582	22.05		.0925
66400	1.5569	.05203	19.219	216.66	573.58	.7519	.8671	.06921	.2631	3.8013	16.665	22.16	.790	.0921
66500	1.5495	.05179	19.308	216.66	573.58	.7519	.8671	.06888	.2624	3.8104	16.742	22.26		.0917
66600	1.5421	.05154	19.402	216.66	573.58	.7519	.8671	.06855	.2618	3.8195	16.823	22.37	.790	.0912
66700	1.5347	.05129	19.497	216.66	573.58	.7519	.8671	.06822	.2612	3.8287	16.905	22.48		.0908
66800	1.5273	.05104	19.592	216.66	573.58	.7519	.8671	.06789	.2606	3.8380	16.988	22.59	.790	.0903
66900	1.5200	.05080	19.685	216.66	573.58	.7519	.8671	.06756	.2599	3.8472	17.068	22.70		.0899



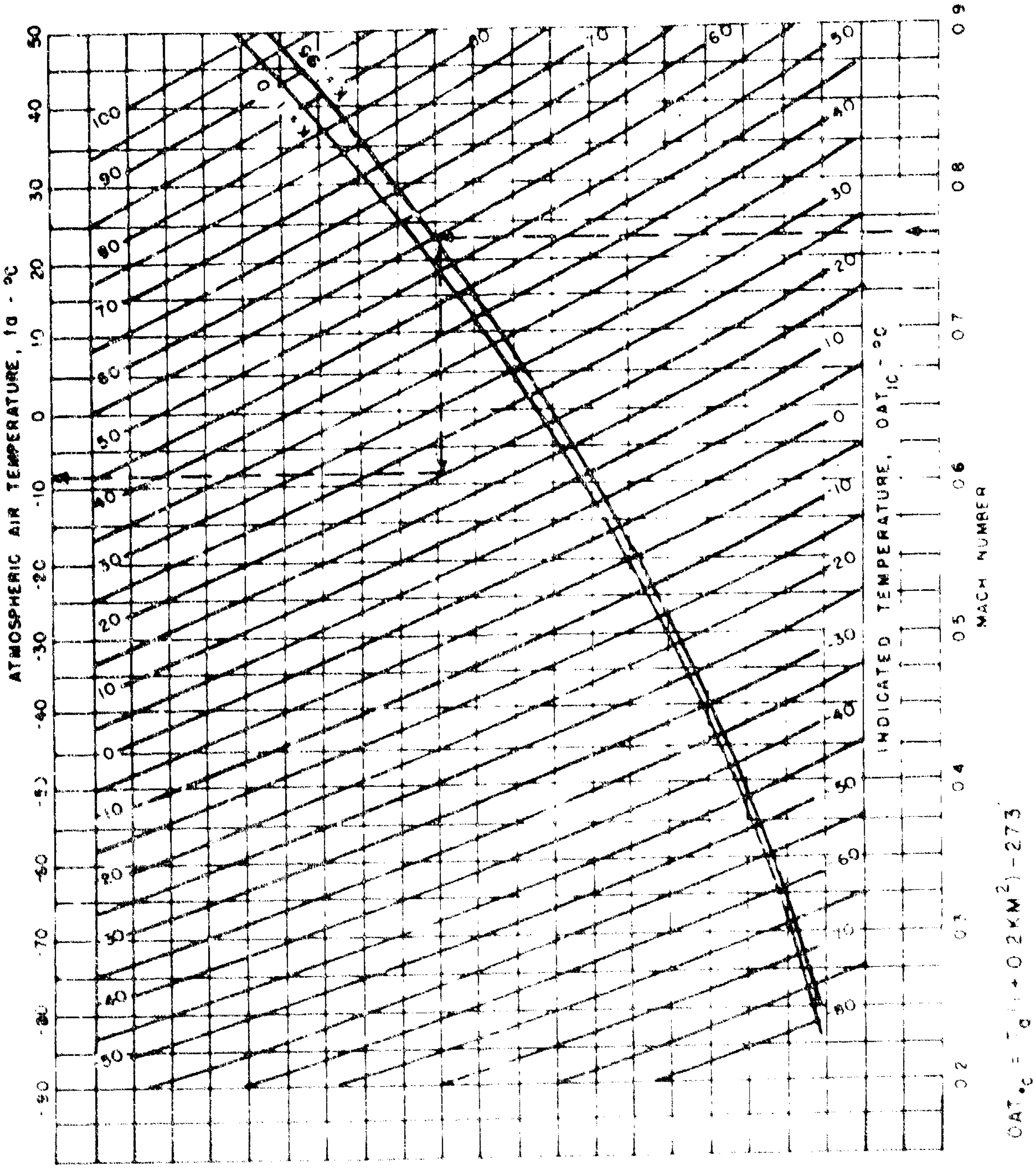
COMPRESSIBILITY CORRECTION

$$V_e = V_c + \Delta V_c$$

AIRSPEED - ALTITUDE - MACH NUMBER

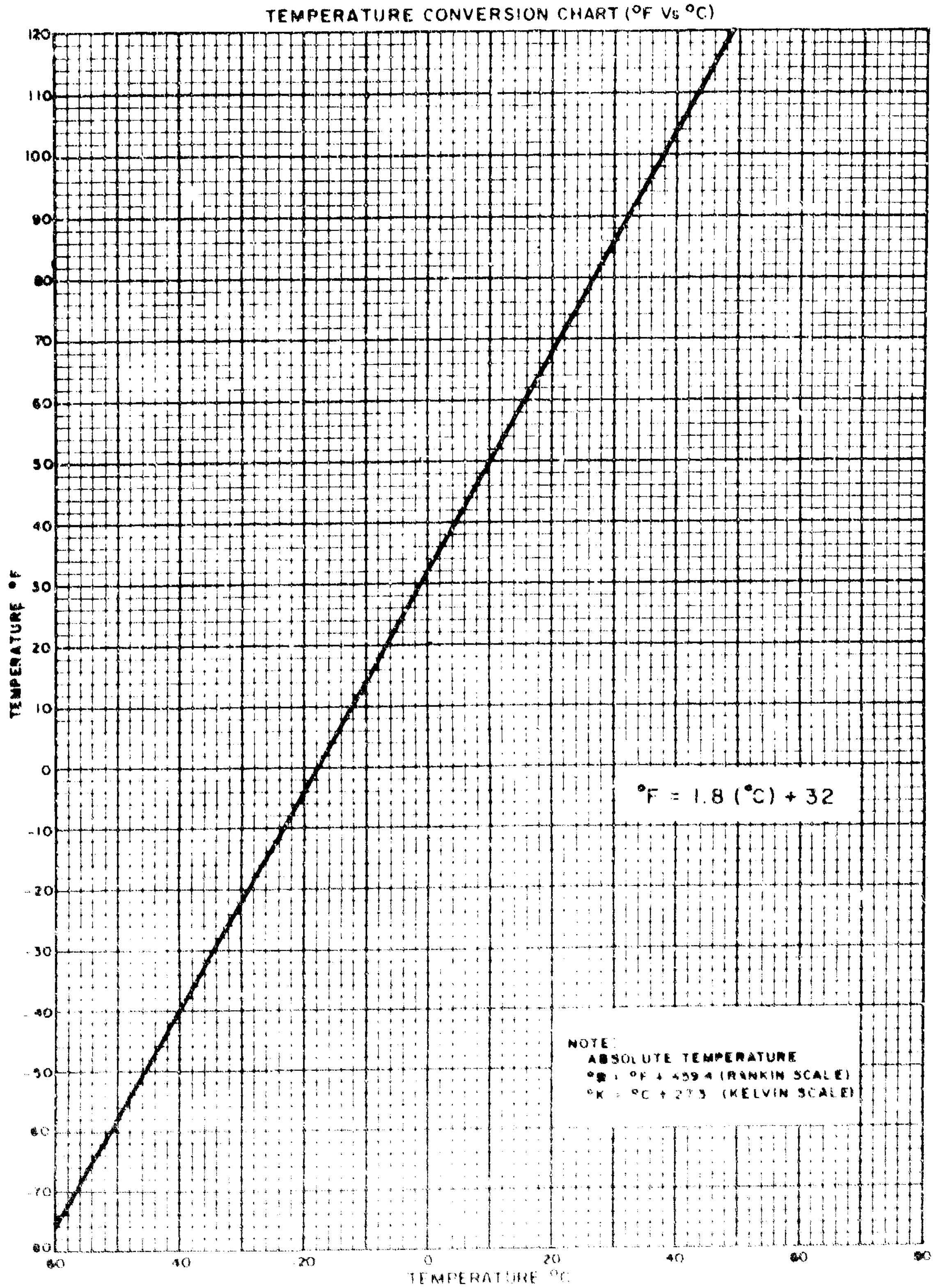


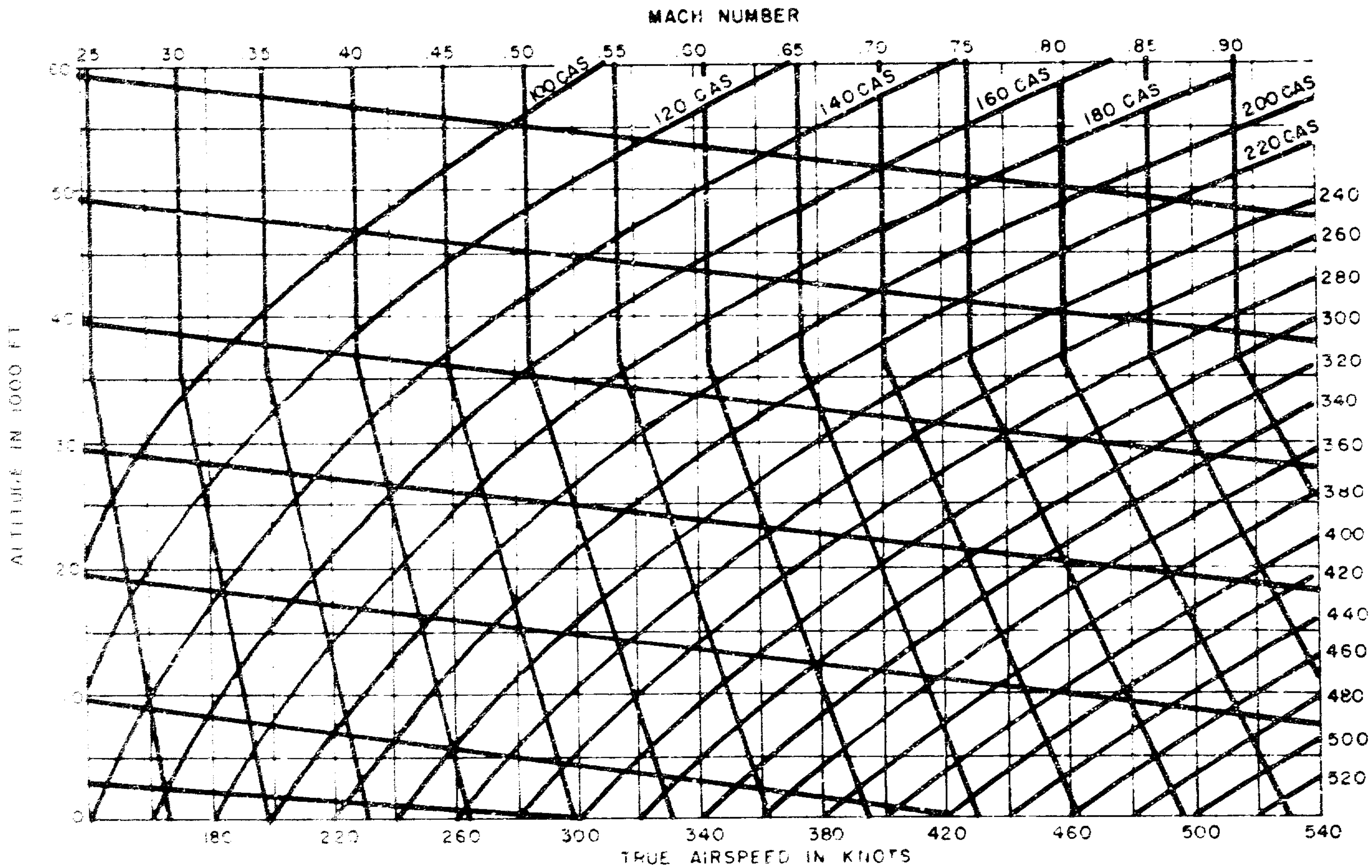
$$M = \left\{ \frac{2}{\gamma - 1} \left[\left(\frac{q_c}{P_a} + 1 \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] \right\}^{1/2}$$



ATMOSPHERIC AIR TEMPERATURE DETERMINATION

TEMPERATURE CONVERSION CHART (^{O}F Vs ^{O}C)





AIRSPEED RELATIONS CALIBRATED AIRSPEED
(ICAO STANDARD ATMOSPHERE)

STATIC PRESSURE - PRESSURE ALTITUDE
(psf)

PRESSURE ALTITUDE Hp. Feet	00	10	20	30	40	50	60	70	80	90
-1000	2193.8	2194.6	2195.4	2196.2	2197.0	2197.8	2198.5	2199.3	2200.1	2200.9
-900	2186.0	2186.7	2187.5	2188.3	2189.1	2189.9	2190.7	2191.5	2192.2	2193.0
-800	2178.1	2178.9	2179.7	2180.5	2181.2	2182.0	2182.8	2183.6	2184.4	2185.2
-700	2170.3	2171.1	2171.9	2172.6	2173.4	2174.2	2175.0	2175.8	2176.6	2177.3
-600	[illegible]	2163.3	2164.1	2164.8	2165.6	2166.4	2167.2	2168.0	2168.7	2169.5
-500	2154.7	2155.5	2156.3	2157.1	2157.8	2158.6	2159.4	2160.2	2160.9	2161.7
-400	2147.0	2147.8	2148.5	2149.3	2150.1	2150.9	2151.6	2152.4	2153.2	2154.0
-300	2139.3	2140.0	2140.8	2141.6	2142.3	2143.1	2143.9	2144.7	2145.4	2146.2
-200	2131.6	2132.3	2133.1	2133.9	2134.6	2135.4	2136.2	2136.9	2137.7	2138.5
-100	2123.9	2124.6	2125.4	2126.2	2126.9	2127.7	2128.5	2129.2	2130.0	2130.8
0	2116.2	2117.0	2117.7	2118.5	2119.3	2120.0	2120.8	2121.6	2122.3	2123.1
0	2116.2	2115.5	2114.7	2113.9	2113.2	2112.4	2111.6	2110.9	2110.1	2109.3
100	2108.6	2107.8	2107.1	2106.3	2105.5	2104.8	2104.0	2103.2	2102.5	2101.7
200	2101.0	2100.2	2099.4	2098.7	2097.9	2097.2	2096.4	2095.6	2094.9	2094.1
300	2093.4	2092.6	2091.9	2091.1	2090.3	2089.6	2088.8	2088.1	2087.3	2086.6
400	2085.8	2085.0	2084.3	2083.5	2082.8	2082.0	2081.3	2080.5	2079.8	2079.0
500	2078.3	2077.5	2076.8	2076.0	2075.2	2074.5	2073.7	2073.0	2072.2	2071.5
600	2070.7	2070.0	2069.2	2068.5	2067.7	2067.0	2066.2	2065.5	2064.7	2064.0
700	2063.2	2062.5	2061.7	2061.0	2060.2	2059.5	2058.7	2058.0	2057.2	2056.5
800	2055.7	2055.0	2054.3	2053.5	2052.8	2052.0	2051.3	2050.5	2049.8	2049.0
900	2048.3	2047.5	2046.8	2046.1	2045.3	2044.6	2043.8	2043.1	2042.3	2041.6
1000	2040.9	2040.1	2039.4	2038.6	2037.9	2037.1	2036.4	2035.7	2034.9	2034.2
1100	2033.4	2032.7	2032.0	2031.2	2030.5	2029.7	2029.0	2028.3	2027.5	2026.8
1200	[illegible]	2025.3	2024.6	2023.8	2023.1	2022.4	2021.6	2020.9	2020.1	2019.4
1300	2018.7	2017.9	2017.2	2016.5	2015.7	2015.0	2014.3	2013.5	2012.8	2012.1
1400	2011.3	2010.6	2009.9	2009.1	2008.4	2007.7	2006.9	2006.2	2005.5	2004.7

STATIC PRESSURE - PRESSURE ALTITUDE
(psf)

PRESSURE ALTITUDE Geo. Feet	00	10	20	30	40	50	60	70	80	90
1500	2004.0	2003.3	2002.5	2001.8	2001.1	2000.3	1999.6	1998.9	1998.1	1997.4
1600	1996.7	1996.0	1995.2	1994.5	1993.8	1993.0	1992.3	1991.6	1990.9	1990.1
1700	1989.4	1988.7	1987.9	1987.2	1986.5	1985.8	1985.0	1984.3	1983.6	1982.9
1800	1982.1	1981.4	1980.7	1980.0	1979.2	1978.5	1977.8	1977.1	1976.3	1975.6
1900	1974.9	1974.2	1973.5	1972.7	1972.0	1971.3	1970.6	1969.8	1969.1	1968.4
2000	1967.7	1967.0	1966.2	1965.5	1964.8	1964.1	1963.4	1962.6	1961.9	1961.2
2100	1960.5	1959.8	1959.0	1958.3	1957.6	1956.9	1956.2	1955.5	1954.7	1954.0
2200	1953.3	1952.6	1951.9	1951.1	1950.4	1949.7	1949.0	1948.3	1947.6	1946.9
2300	1946.1	1945.4	1944.7	1944.0	1943.3	1942.6	1941.9	1941.1	1940.4	1939.7
2400	1939.0	1938.3	1937.6	1936.9	1936.2	1935.4	1934.7	1934.0	1933.3	1932.6
2500	1931.9	1931.2	1930.5	1929.8	1929.1	1928.3	1927.6	1926.9	1926.2	1925.5
2600	1924.8	1924.1	1923.4	1922.7	1922.0	1921.3	1920.6	1919.8	1919.1	1918.4
2700	1917.7	1917.0	1916.3	1915.6	1914.9	1914.2	1913.5	1912.8	1912.1	1911.4
2800	1910.7	1910.0	1909.3	1908.6	1907.9	1907.2	1906.5	1905.8	1905.1	1904.3
2900	1903.6	1902.9	1902.2	1901.5	1900.8	1900.1	1899.4	1898.7	1898.0	1897.3
3000	1896.6	1895.9	1895.2	1894.5	1893.8	1893.1	1892.4	1891.7	1891.0	1890.3
3100	1889.7	1889.0	1888.3	1887.6	1886.9	1886.2	1885.5	1884.8	1884.1	1883.4
3200	1882.7	1882.0	1881.3	1880.6	1879.9	1879.2	1878.5	1877.8	1877.1	1876.4
3300	1875.7	1875.0	1874.4	1873.7	1873.0	1872.3	1871.6	1870.9	1870.2	1869.5
3400	1868.8	1868.1	1867.4	1866.7	1866.0	1865.4	1864.7	1864.0	1863.3	1862.6
3500	1861.9	1861.2	1860.5	1859.8	1859.2	1858.5	1857.8	1857.1	1856.4	1855.7
3600	1855.0	1854.3	1853.6	1853.0	1852.3	1851.6	1850.9	1850.2	1849.5	1848.8
3700	1848.2	1847.5	1846.8	1846.1	1845.4	1844.7	1844.1	1843.4	1842.7	1842.0
3800	1841.3	1840.6	1840.0	1839.3	1838.6	1837.9	1837.2	1836.5	1835.9	1835.2
3900	1834.5	1833.8	1833.1	1832.5	1831.8	1831.1	1830.4	1829.7	1829.1	1828.4

IMPACT PRESSURE (q_c) - CALIBRATED AIRSPEED

CALIBRATED AIRSPEED V_c, Knots	0	1	2	3	4	5	6	7	8	9
50	8.48	8.82	9.17	9.53	9.89	10.26	10.64	11.02	11.41	11.80
60	12.21	12.62	13.04	13.47	13.90	14.34	14.78	15.24	15.70	16.16
70	16.64	17.12	17.60	18.10	18.60	19.10	19.62	20.14	20.67	21.20
80	21.75	22.30	22.85	23.42	23.98	24.56	25.15	25.74	26.33	26.94
90	27.55	28.17	28.79	29.43	30.07	30.71	31.37	32.03	32.69	33.37
100	34.05	34.74	35.43	36.14	36.85	37.56	38.29	39.02	39.75	40.50
110	41.25	42 01	42.77	43.55	44.33	45.11	45.91	46.71	47.52	48.33
120	49.15	49.98	50.82	51.66	52.52	53.37	54.24	55.11	55.99	56.88
130	57.77	58.67	59.58	60.50	61.42	62.35	63.28	64.23	65.18	66.14
140	67.10	68 08	69.06	70.04	71.04	72.04	73.05	74.67	75.09	76.12
150	77.16	78.21	79.26	80.32	81.39	82.46	83.54	84.63	85.73	86.83
160	87.95	89.07	90.19	91.33	92.47	93.61	94.77	95.93	97.11	98.28
170	99.47	100.66	101.86	103.07	104.29	105.51	106.74	107.99	109.22	110.49
180	111.74	113.01	114.28	115.57	116.86	118.16	119.46	120.77	122.10	123.42
190	124.76	126.11	127.46	128.82	130.18	131.56	132.94	134.33	135.73	137.13
200	[illegible]	[illegible]	141.40	142.83	144.28	145.73	147.19	148.65	150.13	151.61
210	153.10	154.60	156.11	157.62	159.14	160.67	162.21	163.76	165.31	166.87
220	168.44	170.02	[illegible]	173.20	174.80	176.41	178.03	179.65	181.29	182.93
230	184.58	186.23	187.90	189.57	191.25	192.94	194.64	196.35	198.06	199.78
240	201.51	203.25	205.00	206.75	208.51	210.28	212.06	213.85	215.65	217.45
250	219.62	221.08	222.91	224.75	226.59	228.45	230.31	232.18	234.06	235.95
260	237.84	239.75	241.66	243.58	245.51	247.44	249.39	251.34	253.31	255.28
270	257.26	259.25	261.25	263.25	265.27	267.29	269.32	271.36	273.41	275.47
280	277.53	279.61	281.69	283.78	285.88	287.99	290.11	292.24	294.37	296.52
290	298.67	300.83	303.01	305.19	307.37	309.57	311.78	313.99	316.22	318.45

IMPACT PRESSURE (q_c) - CALIBRATED AIRSPEED

CALIBRATED AIRSPEED V_c, knots	0	1	2	3	4	5	6	7	8	9
300	320.69	322.94	325.20	327.47	329.75	332.04	334.34	336.64	338.96	341.28
310	343.61	345.94	348.30	350.66	353.03	355.41	357.80	360.20	362.60	365.02
320	367.44	369.88	372.32	374.32	377.23	379.71	332.19	384.68	387.18	389.68
330	392.20	394.73	397.27	399.81	402.37	404.94	407.51	410.10	412.69	415.29
340	417.91	420.55	423.17	425.81	428.46	431.12	433.79	436.48	439.17	441.87
350	444.58	447.30	450.03	452.77	455.52	458.28	461.05	463.83	466.62	469.42
360	472.23	475.05	477.88	480.72	483.57	486.43	489.30	492.18	495.07	497.97
370	500.88	503.80	506.73	509.67	512.63	515.59	518.56	521.54	524.54	527.24
380	530.55	533.58	536.61	539.66	542.71	545.78	548.85	551.94	555.04	558.15
390	561.26	564.39	567.53	570.68	573.84	577.85	580.20	583.39	586.60	589.81
400	593.04	596.27	599.52	602.07	606.05	609.33	612.62	615.92	619.24	622.56
410	625.89	629.24	632.60	635.97	639.35	642.74	646.14	649.55	652.98	656.41
420	659.86	663.32	666.79	670.27	673.76	677.26	680.78	684.30	687.84	691.39
430	694.95	698.52	702.70	705.70	709.30	712.92	716.55	720.20	723.85	727.52
440	731.19	734.88	738.58	742.29	746.02	749.75	753.50	757.26	761.03	764.82
450	768.61	772.42	776.24	780.07	783.92	787.77	791.64	795.52	799.41	803.32
460	807.24	811.17	815.11	819.06	823.03	827.01	831.00	835.00	839.02	843.05
470	847.09	851.14	855.21	859.29	863.38	867.48	871.60	875.73	879.87	884.03
480	888.20	892.38	896.57	900.78	905.00	909.23	913.48	917.74	922.01	926.29
490	930.59	934.90	939.23	943.56	947.91	952.28	956.66	961.05	965.45	969.87
500	974.50	978.74	983.20	987.67	992.15	996.65	1001.16	1005.69	1010.23	1014.78
510	1018.55	1023.93	1028.52	1033.13	1037.75	1042.38	1047.03	1051.69	1056.37	1061.06
520	1065.77	1070.49	1075.22	1079.97	1084.73	1089.50	1094.29	1099.10	1103.91	1108.75
530	1113.59	1118.46	1123.35	1128.22	1133.12	1138.04	1142.98	1147.92	1152.89	1157.87
540	1162.56	1167.66	1172.88	1177.92	1182.97	1188.04	1193.12	1198.21	1203.32	1208.45

IMPACT PRESSURE (q_c) - CALIBRATED AIRSPEED

CALIBRATED AIRSPEED V_c, Knots	0	1	2	3	4	5	6	7	8	9
550	1213.59	1218.75	1223.92	1229.10	1234.30	1239.52	1244.75	1250.00	1255.26	1260.54
560	1265.83	1271.14	1276.46	1281.80	1287.15	1292.52	1297.91	1303.31	1308.73	1314.16
570	1319.61	1325.07	1330.55	1336.05	1341.56	1347.08	1352.63	1358.19	1363.76	1369.35
580	1374.96	1380.58	1386.22	1391.88	1397.55	1403.24	1408.94	1414.67	1420.40	1426.16
590	1431.93	1437.71	1443.52	1449.34	1455.17	1461.03	1466.90	1472.78	1478.69	1484.61
600	1490.55	1496.50	1502.47	1508.47	1514.47	1520.49	1526.53	1532.58	1538.66	1544.75
610	1550.86	1556.98	1563.13	1569.29	1575.46	1581.66	1587.87	1594.10	1600.35	1606.62
620	1612.90	1619.20	1625.52	1631.86	1638.21	1644.58	1650.97	1657.38	1663.81	1670.25
630	1676.71	1683.19	1689.69	1696.21	1702.75	1709.30	1715.87	1722.46	1729.07	1735.70
640	1742.35	1749.01	1755.69	1762.40	1769.12	1775.86	1782.61	1789.39	1796.19	1803.00
650	1809.84	1816.69	1823.56	1830.45	1837.36	1844.29	1851.24	1858.21	1865.20	1872.21
660	1879.23	1886.28	1893.35	1900.43	1907.54	1914.66	1921.80	1928.97	1936.15	1943.35
670	1950.57	1957.81	1965.07	1972.35	1979.64	1986.96	1994.29	2001.65	2009.02	2016.41
680	2023.82	2031.24	2038.69	2046.15	2053.64	2061.14	2068.66	2076.20	2083.75	2091.33
690	2098.92	2106.53	2114.16	2121.81	2129.47	2137.15	2144.85	2152.57	2160.31	2168.06
700	2175.83	2183.62	2191.43	2199.25	2207.09	2214.95	2222.83	2230.73	2238.64	2246.57
710	2254.51	2262.48	2270.46	2278.45	2286.47	2294.50	2302.55	2310.62	2318.70	2326.80
720	2334.92	2343.06	2351.21	2359.38	2367.56	2375.76	2383.98	2392.22	2400.47	2408.74
730	2417.03	2425.33	2433.65	2441.98	2450.33	2458.70	2467.09	2475.49	2483.91	2492.34
740	2500.79	2509.26	2517.74	2526.24	2534.75	2543.29	2551.83	2560.40	2568.98	2577.57